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A Direct Synthesis Method for the Conceptual Design of Transport Aircraft

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A DIRECT SYNTHESIS METHOD FOR THE CONCEPTUAL DESIGN OF TRANSPORT AIRCRAFT

by

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Abstract

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The problem of synthesizing a transport aircraft at the conceptual design level is considered. A direct sizing algorithm that does not require iteration is developed. Such direct synthesis methods can be used as important building blocks in an aircraft optimization process. New statistical equations based on current aircraft are derived for approximating the widths and lengths of the cabin and fuselage. A more accurate static thrust over gross weight, which is based on the equations of motion specified by the FAR part 25 climb requirements, is presented. A cruise at constant altitude with optional step-climb is taken into account. The operating empty weight is directly estimated using a statistical correlation for current twin turbofan transport aircraft. The analyses of the layout, performance, and weights are finally assembled in a direct sequential algorithm. This design method is tested on an existing transport aircraft and the results agreed well with the actual design.

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List of Symbols

- a : speed of sound
- a_c : centripetal acceleration
- a_t : tangential acceleration
- A : wing aspect ratio
- c_a : acceleration constant for climb
- $c_v = -.949(10)^{-3}\frac{lb}{ft/s^2}$
- $c_{v^2} = .773(10)^{-6}\frac{lb}{(ft/s^2)^2}$
- C_D : drag coefficient
- C_{D_o} : clean configuration parasite drag coefficient
- $C_i = \frac{2W^2}{\rho S\pi Ae}$
- C_L : lift coefficient
- $C_{L_{max}}$: maximum lift coefficient
- $C_p = \frac{1}{2}\rho SC_{D_o}$
- C_{TSFC} : thrust specific fuel consumption
- d : constant braking deceleration

- d_e : emergency exit dimension
- d_f : front distance
- d_r : rear distance
- D : drag
- $D_{induced}$: induced drag
- $D_{parasite}$: parasite drag
- ΔC_{D_o} : increment of parasite drag coefficient
- Δn : load factor increment
- Δt_{fgr} : free ground roll duration
- e : clean configuration Oswald efficiency factor
- E : endurance
- ϕ : angle between thrustline and flight path
- $g = 32.2$ ft/sec^2
- γ : approach, climb or descent angle
- h : altitude
- h_{af} : approach-flare transition altitude
- h_c : cruising altitude
- h_{oc} : optimal cruising altitude
- l_a : approach segment
- l_b : braking segment

- l_c : cabin length
- l_f : fuselage length
- l_{fgr} : free ground roll segment distance
- l_{f1} : first flare segment distance
- l_{f2} : second flare segment distance
- l_{gr} : ground roll segment distance
- l_{nt} : fuselage nose-tail cone length
- L : lift
- LFL : FAR part 25 landing field length
- M : Mach number
- M_c : cruising Mach number
- M_{oc} : optimal cruising Mach number
- $M_{oc@sl}$: equivalent optimal cruising Mach number at sea level
- n : number of operating engines
- n_c : number of crew members
- n_e : number of economy class passengers
- n_{ea} : number of economy class passengers per flight attendant
- n_f : number of first class passengers
- n_{fa} : number of first class passengers per flight attendant
- n_T : number of engines

- nsa_e : number of seats abreast in economy class
- nsa_f : number of seats abreast in first class
- p : air pressure
- p_e : seat pitch in economy class
- p_f : seat pitch in first class
- p_{oc} : optimal cruise air pressure
- p_{sl} : sea level air pressure
- $\pi = 3.141592$
- r : radius of the circular flare arc
- r_{nt} : fuselage nose-tail cone ratio
- R : range
- ρ : air density
- ρ_{sl} : air density at sea level
- ρ_{oc} : air density at optimal cruising altitude
- s_{fe} : first/economy class separation distance
- S : wing reference area
- t : time
- t_f : fuselage thickness
- T : thrust or air temperature
- T_{oc} : optimal cruise air temperature

- T_s : static thrust
- T_{sl} : sea level air temperature
- $T_{s/e}$: static thrust per engine
- V : speed
- V_a : approach speed
- V_b : braking speed
- V_{bc} : speed at the beginning of the cruise
- V_c : cruising speed
- V_f : flare speed
- V_{fgr} : free ground roll speed
- V_{oc} : optimal cruising speed
- $V_{oc@sl}$: optimal cruising equivalent speed
- V_s : stall speed
- $V_{s@l}$: stall speed in landing configuration
- $V_{s@t}$: stall speed in takeoff configuration
- w_{ae} : aisle width in economy class
- w_{af} : aisle width in first class
- w_c : cabin width
- w_f : fuselage width
- w_{saf} : average seat and aisle width in first class

- w_{se} : seat width in economy class
- w_{sf} : seat width in first class
- W : weight
- W_a : weight per flight attendant
- W_{bc} : weight at the beginning of the cruise
- W_{bl} : weight at the beginning of the loiter
- W_c : cargo weight
- W_e : operating empty weight
- W_{ec} : weight at the end of the cruise
- W_{el} : weight at the end of the loiter
- W_{ep} : weight per economy class passenger
- W_f : fuel weight
- W_{fc} : weight per flight crew member
- W_{fp} : weight per first class passenger
- W_g : gross weight (maximum ramp weight)
- W_{ml} : maximum landing weight
- W_p : crew and payload weight
- x_b : stall speed fraction for braking
- x_{cl} : climb weight fraction
- $x_{cr\&aa}$: cruise and diversion to alternate airport weight fraction

- $x_{C_{L_{max}}}$: $C_{L_{max}}$ fraction for pertinent configuration
- $x_{C_{L_{max}}@t}$: $C_{L_{max}}$ fraction for takeoff configuration
- x_{de} : descent weight fraction
- x_e : Oswald efficiency fraction
- x_f : stall speed fraction for flare
- x_{fgr} : stall speed fraction for free ground roll
- x_{fu} : fuel weight fraction
- x_{la} : landing, taxi and shut-off weight fraction
- x_{lo} : loiter weight fraction
- x_{ml} : maximum landing weight over gross weight
- x_{mt} : maximum takeoff weight over gross weight
- x_t : instantaneous thrust fraction
- x_{to} : start, warm-up, taxi and takeoff weight fraction
- x_{uf} : unusable fuel weight fraction
- x_v : instantaneous stall speed fraction
- x_w : instantaneous weight fraction

Chapter 1

Introduction

This thesis is part of an ongoing research project to optimize the synthesis of aircraft at the conceptual design level. This type of aircraft design problems has a mixture of continuous and discrete variables. For example, if the synthesis problem of high capacity transport aircraft is considered, some of the possible continuous variables are the wing aspect ratio, taper ratio and sweep angle, whereas the number of decks, numbers of seats abreast in each deck are examples of discrete variables. Traditional methods of optimization, such as the conjugate gradient method, are not suited well for this type of problem. A promising method that has been attempted recently is the genetic algorithm. Preliminary studies with this technique have shown that it is preferable to have an optimization algorithm that is a direct sequential rather than iterative synthesis. For example, it was found that if an iterative method is used, some convergence problems may be encountered especially for 'unusual' combinations of the design variables. Therefore, the purpose of this thesis is to develop a direct (i.e. no iteration) and fast synthesis method that can be readily incorporated to an optimization method such as genetic algorithm or other promising algorithms that might become available in the future.

Once the direct constraining equations are obtained, they can be added to the optimization problem. For expediency, current twin-engine transport aircraft data were gathered to obtain a statistical criterion for selecting the best design.

In the framework of this master's thesis, the present text deals only with the conceptual design algorithm of transport aircraft without the optimization module. However, the computer programs that have been developed as a result of the present research are included in the appendices for the benefit of the readers.

A conceptual design consists of determining the most basic characteristics of a transport aircraft which satisfies a given set of design requirements. Reference books in the area of aircraft design are frequently structured as a guide to procedures where choices based on designers'experience are required throughout the process. In general, correlations and statistical expressions are extensively used for getting important aircraft characteristics. However, these references have not been published or revised recently so their statistical estimations may be inaccurate for current technology aircraft where, for example, the impact of modern propulsion systems and the use of advanced material might be important. Usually, it is the choice of the designer to take into account specific regulations that are inherent to the type of aircraft to be synthesized.

For the present study, the statement for the conceptual design problem is reformulated in a way that can be simplified while getting detailed and reliable aircraft characteristics. Whereas the parameters representing the design requirements can be assumed to be fixed, one may wish to study the effect of varying a set of variables of designer's choices. This is especially important in sensitivity and optimization studies. Aircraft characteristics that are similar over a large population of current airplanes may be used to constrain, once and for all, some of the designer's variables, provided the design concept is similar to those aircraft used in the design database.

Such decisions can later be validated while testing the design equations with existing aircraft. Several important certification constraints usually not explicitly mentioned among the requirements can be imposed at this early stage of the design to help generate realistic configurations.

The second chapter introduces the reader to the specific aircraft conceptual design problem at hand: the design requirements and the designer's options (input) for the concept and the results (output) of the conceptual design algorithm. Subsequent chapters deal with the different equations of the algorithm. In chapter three, the cabin and fuselage sizing are based on the seating arrangement and passenger capacity. In chapter four, the stall speed in landing configuration is calculated from the landing procedure. In chapter five, the stall speed definition is used to calculate the stall speed in takeoff configuration. In chapter six, the wing-loading is obtained from the stall speed definition at maximum landing weight. In chapter seven, the required static thrust over gross weight is estimated from the stall speed that meets the critical climb requirement. In chapters eight and nine, the important cruise and loiter weight fractions are studied with respect to the design requirements. In chapter ten, the payload weight is determined from the passenger capacity while the operating empty weight is statistically estimated from known aircraft parameters. In the same chapter, the fuel weight is calculated from weight fractions of the different mission segments including cruise and loiter weight fractions. Chapter eleven presents the design synthesis algorithm based on the equations presented separately in the preceding chapters. In order to validate the method, an existing aircraft is 'redesigned' from some of its known performance characteristics and the result is compared to the remaining data for the aircraft.

Chapter 2

Conceptual Design Problem

The conceptual design problem for a transport aircraft is presented first. It is divided into three design categories : requirements, data and results. The notation of the relevant variables is also introduced.

2.1 Requirements

The conceptual design is to fullfill a list of transport aircraft requirements corresponding to the prospective needs of the airlines. These constraints are usually established on the basis of market surveys involving major airline operators. The following list of design specifications is based on a request for proposal for a student design competition available in [AIAA] concerning the design of a transport aircraft.

- Number of first class passengers : n_f
- Number of economy class passengers : n_e
- Cruising altitude : h_c
- Cruising Mach number : M_c
- Range : R

plus either

- FAR part 25 landing field length : LFL
- Constant braking deceleration : d

or

- Approach speed : V_a

The number of passengers has been divided between first and economy class passengers in order to improve the accuracy of the cabin sizing. The cruising altitude is specified since it is usually assigned by air traffic controllers with only a minimum regard to the aircraft optimal cruising condition. The design range is stated for the design number of passengers with luggage and no other cargo such as containers, pallets, etc.

Two alternatives have been considered for the design problem: either the FAR part 25 landing field length and the constant braking deceleration are specified or the approach speed is imposed. The first choice is more likely to represent actual design requirements. Indeed, the field length may be a criterion for landing at some airports. In addition, the airlines may also require a constant braking deceleration to insure passenger comfort. Even though these two problem statements do not appear to be related, the FAR part 25 landing field length and the constant braking deceleration are sufficient to determine the approach speed. Even if the approach speed is specified, this method is still applicable.

The FAR part 25 takeoff field length criterion is not considered at this stage, since its use would make the aircraft design process an undesirable iterative one. Indeed, for optimization problems, a direct process is preferred. However, this criterion should be checked before the design is completed. For current transport aircraft, it seems that the takeoff is usually not the most critical constraint. So, if the takeoff field length is within current average margins, this criterion should be satisfied automatically.

2.2 Design Data

On the basis of experience, the designer should provide some preliminary estimates of the concept in order to initiate the design algorithm. For the present analysis, the set of designer's choice is:

- Number of seats abreast in economy class : nsa_e
- Wing aspect ratio : A
- Maximum lift coefficient : $C_{L_{max}}$
- Maximum landing over gross weight : x_{ml}

The number of seats abreast and the wing aspect ratio respectively characterize the fuselage and wing shapes, respectively. The maximum lift coefficient depends upon the type of high-lift devices to be used. The landing weight fraction implicitly sizes the landing gear: strength and number of wheels. If the designer is not satisfied with the outcome of the design, the values of the design variables can be modified in order to improve the sizing. An optimization algorithm can also be used to improve the efficiency of this process.

2.3 Design Results

On the basis of the designer's choices, the algorithm evaluates the remaining important conceptual parameters that are to satisfy the set of proposed design requirements. The results of this design process are the following:

- Number of seats abreast in first class : nsa_f

- Cabin width : w_c
- Cabin length : l_c
- Fuselage width : w_f
- Fuselage length : l_f
- Wing loading : W_g/S
- Static thrust over gross weight : T_s/W_g
- Operating empty weight : W_e
- Crew and payload weight : W_p
- Fuel weight : W_f

The number of passengers and seats abreast in first and economy classes establish the seating arrangement and, thus, the cabin width and length. The fuel weight that is necessary to cover the design range with the desired passenger capacity, allows the calculation of the gross weight from the operating empty weight as well as the payload and crew weight. Two pilot cockpit crew are taken into account in the estimation of the crew weight. The number of flight attendents is determined from the number of passengers in first and economy classes. The wing area and static thrust can be calculated from the wing loading and the static thrust over gross weight, respectively. Based on current technology turbofans, the engines will be characterized by their static thrust only. The span is determined from the desired wing aspect ratio and the calculated wing planform area. Since the maximum lift coefficient has also been selected, and since the sweep angle of transport aircraft does not vary much from 30 degrees, the wing geometry and the type of high-lift devices become essentially defined.

A conceptual design problem of transport aircraft was defined in this chapter. From now on, different synthesis equations will be developed in the next chapters in order to solve sequentially this particular problem.

Chapter 3

Cabin and Fuselage Sizing

"The fuselage represents such an important item in the total concept that its design might well be started before the overall configuration is settled." [Torenbeek]

The cabin width, fuselage width, cabin length and fuselage length are approximated from the seating arrangement, i.e. the number of seats abreast in economy class, the number of first class passengers and the number of economy class passengers. Only the single deck configuration is considered here.

3.1 Cabin Width

The sizing of the cabin usually starts with the cabin cross-section. Since the shape is often chosen to be nearly circular, this problem reduces to define a cross-section that is big enough to enclose the passengers, their carry-ons and baggages. However, a circular section is in general the desirable shape, since the structure is lighter and easier to build, the pressure load being uniform around the circumference. So, it was decided that the cross-section design would be represented by only one design parameter: the cabin width. Since the drag and weight of the fuselage increase with the cross-sectional area, the cabin width should be kept at a minimum. Thus, one may want to represent this parameter as a discrete function (non-continuous) of the number of seats abreast. Since the proportion of first class with respect to the economy class seating is usually much smaller, the internal cabin diameter is sized from the number of seats abreast in economy class. FAR part 25 states that the number of seats on each side of an aisle is to be limited to three. This implies that, for over seven seats abreast, the aircraft cross section should have two aisles.

3.1.1 Specified Widths

In general, the seat and aisle dimensions are selected within a range of values. In connection with this, some guidelines can be found in [Raymer '92] and [Torenbeek]: in economy class, the seat width varies from 17 to 22 inches, while the aisle width is between 18 and 20 inches. In equation 3.1, the '/' symbol represents a division that results in the greatest whole number, i.e., if nsa_e is less than 7, the result is 0. If it is greater than 7, but less than 14, the result is 1. Then, on the basis of the FAR part 25 regulations:

$$w_c = \frac{w_{se} nsa_e + w_{ae}(1 + (nsa_e/7))}{12} \tag{3.1}$$

where

- w_c : cabin width (ft)
- w_{se} : seat width in economy class (in)
- w_{ae} : aisle width in economy class (in)
- nsa_e : number of seats abreast in economy class

3.1.2 Statistical Widths

Whenever the designer is not certain of the seat and aisle widths, the cabin width may be evaluated using statistics on data available for current transport aircraft. The

Aircraft	w_c (ft)	nsa_e
A320 (3 versions)	12.1	6
A321 (1 version)	12.1	6
B737-300 (3 versions)	11.3	6
B757 (4 versions)	11.6	6

Table 1: w_c and nsa_e for one aisle [Jane's 94-95]

Aircraft	w_c (ft)	nsa_e
A300-600 (2 versions)	17.3	8
A310 (1 version)	17.3	8
A310 (1 version)	17.3	9
B767-200 (4 versions)	15.5	7
B767-200 (2 versions)	15.5	8

Table 2: w_c and nsa_e for two aisles [Jane's 94-95]

analysis is divided into two parts since the cross-section may have one or two aisles. Tables 1 and 2 represent the aircraft from [Jane's 94-95] for which both the cabin width and the number of seats abreast in economy class were given.

Since all the aircraft from Table 1 have six seats abreast, a linear regression can not be applied. Therefore, the average internal diameter over the eleven different versions was calculated and found to be 11.7 feet. The percent difference between this value and the actual data in Table 1 was less than 3%. For this average diameter, the seat width was later computed from the aisle width using equation 3.1. Since the actual aisle width ranges from 18 to 20 inches and the cabin width is 11.7 ft for six seats abreast, the seat width varies between 20.1 and 20.4 inches according to equation 3.1. Thus, it seems that an average seat width of 20.25 inches and an average aisle width of 19 inches should be used in equation 3.1 whenever the cabin has six seats abreast or less.

From the data in Table 2, a linear approximation over those ten different versions gave the following expression for cabins with two aisles:

$$w_c = .966nsa_e + 8.78 \tag{3.2}$$

where the statistical correlation factor was .7 and w_c is obtained in feet.

Equation 3.2 should be interpreted carefully. The slope .966 represents the width increment in feet for every seat abreast added in economy class. The reader should not consider this slope independently as the actual seat width. Also, the y-intercept by itself is not the cumulative width of two aisles, it includes a part of the width of a minimum six seats abreast in economy class. The accuracy of equation 3.2 is statistically better than the approximate seat and aisle widths calculated for the single aisle cross-section. In Table 2, the percent difference between the predicted and actual cabin widths is always smaller than 6.5% with equation 3.2, while the error reaches 12% with equation 3.1, for the average seat and aisle widths of a single aisle cross-section.

3.2 Fuselage Width

According to [Torenbeek] and [Raymer '92], the external cabin diameter is approximately 8 inches larger than the internal diameter and is statistically independent of the aircraft size, therefore:

$$w_f = w_c + \frac{2t_f}{12} \tag{3.3}$$

where

- w_f : fuselage width (ft)
- t_f : fuselage thickness (in)

Aircraft	w_c (ft)	w_f (ft)	$2t_f$ (in)
A300-600	17.3	18.5	7
A310	17.3	18.5	7
A320	12.1	13	5
A340	17.4	18.5	7
B767-200	15.5	16.5	6
B777-200	19.3	20.3	6.5
MD-80	10.3	11.8	9
MD-11	18.8	19.8	6

Table 3: t_f calculated from w_c and w_f [Jane's 94-95]

These estimates should be compared with Table 3 which lists all the transport aircraft in [Jane's 94-95] whose cabin width and fuselage width were given. The fuselage thickness was calculated and is shown also.

3.3 Number of Seats Abreast in First Class

The number of seats abreast in first class is calculated from the cabin width as the maximum number of seats abreast that can fit within the diameter in compliance with the chosen standards for the first class. Using this process, the first class usually enjoys more room than is specified. The number of seats abreast in first class can then be written as equation 3.4. According to [Raymer '92] and [Torenbeek], usual seat and aisle widths both range from 20 to 28 inches. Again, as previously explained, the '/' symbol yields integers only.

$$nsa_f = (12w_c - w_{af}(1 + (nsa_e/7))/w_{sf} \tag{3.4}$$

where

- nsa_f : number of seats abreast in first class
- w_{af} : aisle width in first class (in)

Aircraft	nsa_f	w_c (ft)	nsa_e
A320 (1 version)	4/6	12.1	6
A320 (1 version)	6/6	12.1	6
A321 (1 version)	4/4	12.1	6
B737-300 (1 version)	4/4	11.3	6
B757 (4 versions)	4/4	11.6	6

Table 4: Actual/calculated nsa_f, w_c and nsa_e for one aisle [Jane's 94-95]

Aircraft	nsa_f	w_c (ft)	nsa_e
A300-600 (1 version)	6	17.3	8
A310 (1 version)	6	17.3	8
B767-200 (1 version)	6	15.5	7

Table 5: nsa_f, w_c and nsa_e for two aisles [Jane's 94-95]

- w_{sf} : seat width in first class (in)

Tables 4 and 5 are the list of current aircraft in [Jane's 94-95] whose seating arrangement and cabin width were given. The seat and aisle widths were assumed to be the same, since their range varies over the same interval. The average seat and aisle width can then be calculated from Tables 4 and 5. For single and two aisles cabins, the average widths are 27.16 and 28.63 inches, respectively.

The average widths were succesfully tested with equation 3.5 for the aircraft in Table 4 and 5. The predictions of the number of seats abreast were correct for all the aircraft except the A320 which had a calculated value of six seats abreast in first class instead of an actual value of four.

$$nsa_f = round(\frac{12w_c}{w_{saf}}) - (1 + (nsa_e/7)) \tag{3.5}$$

where w_{saf} : average seat and aisle width in first class (in)

	First class	Economy class
Seat pitch (in.)	38-40	34-36
Passengers/lavatory	10-20	40-60

Table 6: Seat pitches and number of lavatories [Raymer '92], [Torenbeek]

3.4 Cabin Length

3.4.1 Specified Pitches

Once the cross-sectional layout has been fixed, the seat pitch needs to be determined, since most of the cabin length is composed of an discrete number of rows. The seat pitch is the longitudinal distance between the back of two successive seats. The ratio of the number of passengers to the number of seats abreast determines the number of rows, usually a whole number. In addition, the galleys, exit doors and toilets must also be taken into account. Some guidelines from [Raymer '92] and [Torenbeek] are given in Table 6.

[Raymer '92] suggests that for every 10 to 20 rows of passengers there should be a door together with closet space whose length varies between 40 to 60 inches. The number of exit doors depends on the number of passengers according to the FAR. In [Torenbeek], if the aircraft capacity ranges between 140 and 179 passengers, the cabin should include 4 exit doors on each side of the passenger cabin: 2 small (20 by 36 inches) and 2 large (24 by 48 inches) ones. The same source also describes the galley, toilet and wardrobe dimensions for the previous generation of transport aircraft. Small variations in toilet sizes are to be expected. [Raymer '92] and [Torenbeek] suggest an area of about 40 by 40 inches on the deck for toilet facilities. However, the galley and wardrobe dimensions vary significantly from aircraft to aircraft, even in different versions of a same aircraft model. The actual number of exit doors, galleys and toilets in current aircraft are given in Table 7 from [Jane's 94-95].

Aircraft	exits	galleys	toilets
A300-600	8	4	6
A310	6	4	6
A320	6	4	6
A321	6	4	6
B737-300	6	2-4	2-3
B757	8	3	4
B767-200	6	2	5

Table 7: Exit, galley and toilet numbers [Jane's 94-95]

It is important to note that four exit doors correspond to the main doors (two at the front, two at the rear) while the others are emergency exits usually located over the wing. In addition, there is usually a galley near the main doors, i.e., at the front and at the rear of the cabin. According to [Torenbeek], the remaining galleys are typically located at the ends of the cabin in order to easily accommodate varying seating arrangements requested by the airlines. However, they are sometimes located in the middle of the cabin. The wardrobes are usually located adjacent to the galleys.

From the previous references, the cabin length is measured in a series of segments in equation 3.6: front entry space, front galley(s), first class section, separator (with an optional galley), economy class section with emergency exits, back galley(s) and rear entry space. Referring to Table 6, the correct seat pitches for the first class and the economy class should be chosen, if good accuracy is required.

$$l_c = \frac{d_f + p_f(n_f + nsa_f - 1)/nsa_f + s_{fe} + p_e(n_e + nsa_e - 1)/nsa_e + d_e + d_r}{12} \tag{3.6}$$

where

- l_c : cabin length (ft)
- d_f : front distance (in)

- p_f : seat pitch in first class (in)
- n_f : number of first class passengers
- s_{fe} : first and economy section separator distance (in)
- p_e : seat pitch in economy class (in)
- n_e : number of economy class passengers
- d_e : emergency exit dimension (in)
- d_r : rear distance (in)

The front distance is the distance from the front bulkhead of the cabin to the first row of seats in the first class. The emergency exit dimension is the sum of the widths of all the emergency exit doors on one side of the aircraft cabin. The rear distance is the distance from the last row of seats in the economy class to the rear bulkhead of the cabin.

3.4.2 Statistical Method

Equation 3.6 requires a thorough knowledge of the cabin configuration: galleys, wardrobes, toilets, etc. However, references were quite vague concerning this subject. Therefore, it may be difficult to use equation 3.6 at a conceptual design level if this information is not directly available. Table 8 contains the data for the 22 aircraft whose minimum required parameters were given in [Jane's 94-95]: cabin length, seating arrangement and number of passengers in first and economy classes.

Similar to the analysis in Section 3.4.1, the parameters from Table 8 can be combined to form an equation whose coefficients can be determined by the least square method. Since the seat rows constitute most of the cabin space, the number of rows in first and economy class are logical variables. On the other hand, the number of

Aircraft	l_c (ft)	nsa_f	nsa_e	n_f	n_e
A300-600(1)	131.9	6	8	26	240
A300-600(2)	131.9	0	8	0	289
A310(1)	109	6	8	20	200
A310(2)	109	0	9	0	280
A320(1)	89.8	4	6	12	138
A320(2)	89.8	0	6	0	164
A321	112.8	4	6	16	169
B737-300(1)	77.2	4	6	8	120
B737-300(2)	77.2	0	6	0	141
B737-300(3)	77.2	0	6	0	149
B757(1)	118.4	4	6	16	162
B757(2)	118.4	4	6	16	170
B757(3)	118.4	4	6	12	190
B757(4)	118.4	4	6	12	196
B757(5)	118.4	0	6	0	214
B757(6)	118.4	0	6	0	220
B757(7)	118.4	0	6	0	223
B757(8)	118.4	0	6	0	224
B767-200(1)	111.3	6	7	18	198
B767-200(2)	111.3	0	7	0	230
B767-200(3)	111.3	0	7	0	242
B767-200(4)	111.3	0	8	0	255

Table 8: l_c, nsa_f, nsa_e, n_f and n_e [Jane's 94-95]

galleys, toilets, wardrobes, and emergency exits may be assumed to be proportional to the number of passengers. Finally, an aditional term may take into account the main doors, entry aisles, etc. Therefore, the data in Table 8 can be summarized in a linear expression depending on the three parameters: n_f/nsa_f, n_e/nsa_e, $nsa_f + nsa_e$. The results of the multi-linear regression analysis are given in equation 3.7 and the listing of the computer program, sample data and results are included in Appendix A.

$$l_c = .87 + 5.49(n_f/nsa_f) + 2.42(n_e/nsa_e) + .135(n_f + n_e) \qquad (3.7)$$

Again, similar to equation 3.2, the coefficients of equation 3.7 should not be related to the pitch values specified in Section 3.4.1. Equation 3.7 was tested against the data given in Table 8: the maximum discrepancy was 5.4% while the average error was 2.36%. Since the error between the actual and the calculated cabin lengths is small, it is expected that the predicted cabin layout will also satisfy the FAR requirements concerning exit doors.

3.5 Fuselage Length

Whereas the cabin is designed around the passengers, the nose and the tail sections of the fuselage depend mainly on individual manufacturer design practices. According to [Torenbeek], the fuselage nose is between 150% and 200% of the fuselage diameter while the tail cone is 250% to 300% of the fuselage diameter. Thus, the distance that needs to be added to the cabin length to determine the fuselage length is between four and five times the fuselage diameter. This ratio will be called the fuselage nose-tail cone ratio. The latter range of fuselage nose-tail cone ratios leads to large fuselage length variations. For this reason, data were gathered for the transport aircraft in [Jane's 94-95] and are shown in Table 9. Their respective fuselage nose-tail cone ratio was calculated along with their fuselage nose-tail cone length which is the sum of the

Aircraft	l_f (ft)	w_f (ft)	l_c (ft)	r_{nt}	l_{nt} (ft)
A300-600	174.9	18.5	131.9	2.32	43
A310	148	18.5	109	2.11	39
B767-200	155	16.5	111.3	2.65	43.7
B767-300	176	16.5	132.4	2.64	43.6
B777	206	20.3	160.6	2.23	45.4
MD80	135.5	11.8	101	2.92	34.5
MD11	192.4	19.8	152.6	2.01	39.8

Table 9: Calculated r_{nt} and l_{nt} from l_f, w_f and l_c [Jane's 94-95]

nose and tail cone lengths.

$$l_f = l_c + r_{nt}w_f = l_c + l_{nt} \tag{3.8}$$

where

- l_f : fuselage length (ft)
- r_{nt} : fuselage nose-tail cone ratio
- l_{nt} : fuselage nose-tail cone length (ft)

It can be seen from Table 9 that the actual values of the fuselage nose-tail cone ratio vary between two and three rather than four and five as mentioned in [Torenbeek]. On the other hand, the fuselage nose-tail cone length does not vary much for current aircraft. Disregarding the MD80, an average nose-tail ratio and length would be 2.33 and 42.42 ft, respectively, corresponding, in Table 9, to an average percent difference of 7% and 4%, respectively. The nose-tail and cabin length errors are reflected in the fuselage length error. Since the major contributor to the fuselage length is the cabin, the errors from the nose and tail cones are generally not significant.

In this chapter, the widths and lengths of the cabin and fuselage were sized for either specified or statistical dimensions. The statistical analysis was based exclusively on current seating arrangements. The cabin width was expressed as a function of the number of seats abreast in economy class. The fuselage width was computed from the cabin width by adding an average current fuselage thickness. In order to determine the cabin length, the number of seats abreast in first class was required, and it was calculated from the cabin width. The estimated cabin length depends on the number of rows in first and economy classes which are a function of the number of passengers. Finally, average nose and tail section lengths for current transport aircraft were added to the cabin length in order to obtain the fuselage length.

Chapter 4

Stall Speed in Landing Configuration

In this chapter, the landing process is explained in some detail. Then, a constant braking deceleration is calculated from this information. Finally, the stall speed in landing configuration will be obtained from the FAR part 25 landing field length and the constant braking deceleration requirements.

4.1 Landing

The FAR part 25 specifies the landing field length as the horizontal distance the aircraft covers from an altitude of 50 feet above the ground until it comes to a complete stop. Then, FAR part 25 requires that the landing field length be multiplied by a safety factor of five-thirds. Finally, it also states that the approach and touchdown speeds should be greater or equal to 130% and 115% of the stall speed in landing configuration, respectively. For the remainder of this chapter, approach and touchdown speeds are defined as 130% and 115% ,respectively, of the stall speed in landing configuration consistent with [Loftin], [McCormick '79], [McCormick '95], [Raymer '92], [Roskam & Lan] and [Torenbeek]. In order to facilitate the evaluation of the landing field length, the landing process is divided into three segments, i.e., approach, flare and ground roll.

4.1.1 Approach

Consistent with [Raymer '92], [Roskam & Lan] and [Torenbeek], the approach angle will be assumed constant for the approach segment so that the segment length can be obtained from the geometry. The change in altitude between 50 feet height and the approach-flare transition gives the approach segment length for a given approach angle.

$$l_a = \frac{50 - h_{af}}{tan(\gamma)} \tag{4.9}$$

where

- l_a : approach segment length (ft)
- h_{af} : approach-flare transition altitude (ft)
- γ : approach angle

The same references suggest an approach angle of three degrees for jet transport aircraft. This approximation is particularly justified in case of ILS aided landings.

4.1.2 Flare

In [McCormick '79], [McCormick '95], [Raymer '92] and [Roskam & Lan], the flare trajectory is assumed to be a circular arc tangent to both the approach and the ground roll paths. See Figure 10.19 (Flight Path Geometry for Landing Flare) in [Roskam & Lan]. These tangent lines can be extended to inside the flare region where they intersect to bisect the circular arc dividing into two parts.

The first part is the extension of the approach path to the bisector on the ground. It defines a triangle whose height is the approach-flare altitude and the opposite angle is the approach angle. Thus, the length of the first flare segment is:

$$l_{f1} = \frac{h_{af}}{tan(\gamma)} \tag{4.10}$$

where l_{f1} : first flare segment length (ft)

The second part, from the bisector to the beginning of the ground roll, is obtained from the equilibrium of the radial forces. It is observed that the second half of the flare arc angle is equal to half the approach angle. Since this angle is small, its cosine is rounded to unity. The equilibrium of forces along the radial direction is stated in equation 4.11.

$$\frac{W}{g}a_c = L - W \tag{4.11}$$

where

- W : weight (lb)
- $g = 32.2$ ft/s^2
- a_c : centripetal acceleration (ft/s^2)
- L : lift (lb)

Equation 4.11 can be rewritten as equation 4.12 in terms of the load factor and the lift-to-weight ratio. [Raymer '92], [Roskam & Lan], [McCormick '79] and [McCormick '95] proposed a lift-to-weight ratio value of 1.2, while a value of .1 for the load factor increment can be found in [Torenbeek].

$$\frac{a_c}{g} \equiv \Delta n = \frac{L}{W} - 1 \tag{4.12}$$

where Δn is the load factor increment

In [McCormick '79], [McCormick '95], [Raymer '92] and [Roskam & Lan], a constant speed for the flare is assumed. [Raymer '92] takes the average of the approach and touchdown speeds. [Roskam & Lan] mentions 95% of the approach speed. [McCormick '79] and [McCormick '95] prefers to take the approach speed that corresponds to a more conservative choice for estimating the landing field length since it is expected that the fastest touchdown speed will be always slower than the lowest approach speed. The centripetal acceleration for a constant speed circular motion gives equation 4.13.

$$\frac{{V_f}^2}{rg} = \Delta n \tag{4.13}$$

where

- V_f : flare speed (ft/s)
- r : radius of the circular flare arc (ft)

It now appears clearly in equation 4.14 that the circular arc radius depends only on the square of speed :

$$r = \frac{{V_f}^2}{g\Delta n} \tag{4.14}$$

From trigonometry, the second flare sub-segment is obtained as :

$$l_{f2} = \frac{{V_f}^2}{g\Delta n} tan(\frac{\gamma}{2}) \tag{4.15}$$

where l_{f2} : second flare segment length (ft)

And, finally, the flare segment length is the sum of equations 4.10 and 4.15.

$$l_f = \frac{h_{af}}{tan(\gamma)} + \frac{{V_f}^2}{g\Delta n} tan(\frac{\gamma}{2}) \tag{4.16}$$

4.1.3 Ground Roll

The ground roll can be divided into two parts, a free ground roll followed by a braking distance. During the free ground roll, no brakes are applied and the speed is assumed constant. In [McCormick '79] and [McCormick '95], the free ground roll speed is assumed to be equal to the approach speed for two seconds. In [Raymer '92] and [Roskam & Lan], it is assumed to be equal to the touchdown speed for a time delay varying from zero to three seconds. In contrast, [Torenbeek] allows for no free ground roll at all. Equation 4.17 represents the free ground roll distance.

$$l_{fgr} = V_{fgr} \Delta t_{fgr} \tag{4.17}$$

where

- l_{fgr} : free ground roll segment (ft)
- V_{fgr} : free ground roll speed (ft/s)
- Δt_{fgr} : free ground roll duration (s)

The braking distance may seem to be intrinsically difficult to treat due to the simultaneous effects of different braking devices: spoilers, reverse thrust and brakes. A tedious integration of the thrust, drag and friction contributions is undertaken in [Raymer '92] and [Roskam & Lan]. The braking due to reverse thrust is roughly approximated in [Raymer '92] but neglected in [Roskam & Lan] while they both give only a crude estimate for the friction coefficient. On the other hand, an easier approach, based on an assumed constant deceleration, is presented in [McCormick '79], [McCormick '95] and [Torenbeek].

For our problem, it is assumed that onboard computers integrate an automatic braking system for assisting the pilot during the landing roll so that most of his attention is essentially limited to the manual operation of the thrust reversers. When the main landing wheels touch down, a transducer transmits a signal to the spoiler actuators. The delay is less than .2 seconds before the spoilers start extending. When the front landing gear hits the runway, the pilot concentrates on the thrust reversers ; the delay before reverse thrust application varies between 4 to 9 seconds after touchdown and is mainly function of the pilot. For the entire ground roll, an onboard computer prevents skidding and maintains the aircraft deceleration above a selected threshold the pilot has set before landing. When the aircraft speed is high, the spoilers and the thrust reversers are very effective and the brakes are not normally used. Later, a servo-loop controls the pressure applied to the brakes to compensate for decreasing drag. So, except for a very short duration at an an early stage of the landing, the airplane deceleration is kept constant during normal operations.

For a constant deceleration, the energy-work conversion, equation 4.18, states that the initial kinetic energy when the braking is initiated, is equal to the work of the braking forces. In [McCormick '79] and [McCormick '95], the braking is assumed to start at approach speed, while [Torenbeek] prefers the touchdown speed.

$$\frac{1}{2}{V_b}^2 = dl_b \tag{4.18}$$

where

- V_b : braking speed (ft/s)
- d : constant braking deceleration (ft/s^2)
- l_b : braking segment (ft)

Then, equation 4.18 can be rewritten for the braking distance to get equation 4.19.

$$l_b = \frac{V_b^2}{2d} \tag{4.19}$$

The braking deceleration remains undetermined for the moment, but its evaluation is treated in Section 4.3. Finally, the ground roll equation 4.20 is obtained by adding equations 4.17 and 4.19.

$$l_{gr} = V_{fgr}\Delta t_{fgr} + \frac{V_b^2}{2d} \tag{4.20}$$

where l_{gr} : ground roll segment (ft)

4.2 Landing Field Length

The landing field length is defined as the sum of equations 4.9, 4.16 and 4.20. Thus, the FAR part 25 landing field length, LFL, which includes the safety factor is given by equation 4.21.

$$LFL = \frac{5}{3}\left(\frac{50}{tan(\gamma)} + \frac{V_f^2}{g\Delta n}tan(\frac{\gamma}{2}) + V_{fgr}\Delta t_{fgr} + \frac{V_b^2}{2d}\right) \tag{4.21}$$

In equation 4.21, the approach angle, load factor increment and free ground roll duration can be chosen from the references in Sections 4.1.1, 4.1.2 and 4.1.3. Those references also specified the flare, free ground roll, and braking speeds as given fractions of the stall speed in landing configuration. So, regardless of the flare, the landing field length is only a function of the stall speed and the braking deceleration. Thus, if an estimation of this deceleration can be obtained, the landing field length is fully determined from the stall speed in landing configuration.

4.3 Constant Braking Deceleration

According to [McCormick '79] and [McCormick '95], estimates of the braking deceleration are $.41g$ and $.42g$ for B-747-100 and B-767-300-ER, respectively. In [Torenbeek], the deceleration ranges between $.4g$ and $.5g$ for jets with ground spoilers, anti-skid devices and speed brakes. If nosewheel braking is added, the deceleration is between $.5g$ and $.6g$. Note however that although [McCormick '79], [McCormick '95] and [Torenbeek] give close deceleration values, different landing analyses are used as was indicated in Section 4.1.3. So, those estimates can not be compared directly. Moreover, [Torenbeek] does not state which aircraft were used to get this range of deceleration, while the results from [McCormick '79] and [McCormick '95] apply to only two aircraft. In addition, [Torenbeek] states that the average deceleration over 15 unspecified jet transport aircraft is $.37g$. Finally, [Torenbeek] also gives the following: $.55g$ on a dry runway with maximum effort and ignoring the passenger tolerance; $.35g$ on a wet runway with modern anti-skid braking, lift dumpers and reverse thrust, and $.15g$ on a wet runway with simple braking or flooded runway with reverse thrust.

On the other hand, the deceleration can also be calculated on the basis of current aircraft data found in [Jane's 94-95] and the landing field length equation 4.21. This expression can be solved for the constant braking deceleration to get equation 4.22. But, before equation 4.22 can be used, some parameters need to be determined first.

$$d = \frac{{V_b}^2}{2(\frac{3}{5}LFL - \frac{50}{tan(\gamma)} - \frac{{V_f}^2}{g\Delta n}tan(\frac{\gamma}{2}) - V_{fgr}\Delta t_{fgr})} \tag{4.22}$$

Referring to Sections 4.1.1, 4.1.2 and 4.1.3, the angle of approach and the load factor increment were taken to be three degrees and .2, respectively, consistent with the values that appeared in most of the references. The free ground roll is neglected because the trend for transport aircraft design includes an automatic braking systems that are active from touchdown. The braking and flare speeds are set equal to the touchdown and approach speeds, respectively. Thus, just two aircraft parameters are necessary for computing the deceleration: the landing field length and the stall speed in landing configuration.

Table 10 is based on aircraft data from [Jane's 94-95] whose landing field length and approach speed were both given. The reader should be aware that this reference gives a variety of data for each aircraft. For some, only the wet landing field length with partial or full flaps is available. For other aircraft, the landing conditions are not specified, the landing field length may be FAR certified or at maximum landing weight or both. The approach speeds in [Jane's 94-95] are given at basic or maximum landing weight and sometimes without any specification at all. However, all the landing field length given in [Jane's 94-95] seemed to be consistent with respect to known certified FAR landing field lengths for comparable aircraft. So, the whole data set was consequently considered as consisting of FAR part 25 landing field lengths. However, it is important to remember that it is in fact likely that not all the landing field lengths were FAR part 25 landing field lengths, and making this assumption has probably altered the results. A different landing field length definition could very likely have been used in the case of non-American aircraft manufacturers. So, the results included here should be considered as an example for getting an estimation of the constant deceleration. For all the aircraft data in Table 10, the deceleration fators range between .28 and .37 and the mean deceleration is $.33g$. When the B-747 is ignored, the range is from $.31g$ to $.37g$ and the average deceleration for twin engined aircraft becomes $.34g$. For this latter case, the largest deviation is less than 10%. However, it is essential to emphasize that these deceleration values are valid for the landing field length equation 4.21 and should only be interpreted as a trend for current aircraft.

Aircraft	LFL (ft)	V_a (kts)	d (ft/s^2)	% g
A300	5040	135	10.95	.34
A300	5100	136	10.92	.34
A310	4850	135	11.66	.36
A310	5100	135	10.74	.33
B-747	6800	153	9.15	.28
B-757	4630	132	11.99	.37
B-757	4790	132	11.32	.35
B-777	5450	138	10.15	.31
B-777	5600	140	10.04	.31

Table 10: Calculated d and % g from LFL and V_a [Jane's 94-95]

4.4 $V_{s@l}$ Equation

Based on Section 4.3 or other comparable source, a constant braking deceleration can now be selected. The approach angle, the load factor increment, and the duration of the free ground roll can again be chosen on the basis of the recommendations in Sections 4.1.1, 4.1.2 and 4.1.3. Finally, the flare, free ground roll, and braking speeds can also be expressed as a given fraction of the stall speed in landing configuration. Only two parameters remain undetermined: the FAR part 25 landing field length and the stall speed in landing configuration. Equation 4.21 is a second order polynomial with respect the stall speed and its roots can be calculated analytically. For a given landing field length, its positive root, given by equation 4.23, provides an estimate of the stall speed in landing configuration.

$$V_{s@l} = \frac{-x_{fgr}\Delta t_{fgr} + \sqrt{(x_{fgr}\Delta t_{fgr})^2 + 4(\frac{3}{5}LFL - \frac{50}{tan(\gamma)})(\frac{x_f{}^2}{g\Delta n}tan(\frac{\gamma}{2}) + \frac{x_b{}^2}{2d})}}{2(\frac{x_f{}^2}{g\Delta n}tan(\frac{\gamma}{2}) + \frac{x_b{}^2}{2d})} \tag{4.23}$$

where

- $V_{s@l}$: stall speed in landing configuration (ft/s)

- $x_f = \frac{V_f}{V_{s@l}}$: stall speed fraction for flare
- $x_{fgr} = \frac{V_{fgr}}{V_{s@l}}$: stall speed fraction for free ground roll
- $x_b = \frac{V_b}{V_{s@l}}$: stall speed fraction for braking

An important remark concerns the propagated error of the stall speed in landing configuration. The error in the stall speed in landing configuration is half the percent error of the parameters under the square root. From the discussion of Sections 4.1.3 and 4.3, it was explained that the current technology braking starts immediately after the main gear touchdown and, therefore, the free ground roll duration can be neglected. In addition, minimizing the propagated error is another motivation for cancelling the only term which is outside the square root, i.e. the free ground roll term. The simplified expression becomes equation 4.24.

$$V_{s@l} = \sqrt{\frac{\frac{3}{5}LFL - \frac{50}{tan(\gamma)}}{\frac{x_f{}^2}{g\Delta n}tan(\frac{\gamma}{2}) + \frac{x_b{}^2}{2d}}} \tag{4.24}$$

Moreover, for a stated landing field length, the constant braking deceleration drives mainly the stall speed error since the errors coming from approach angle, load factor and both flare and braking speeds should be negligible with respect to the constant deceleration. Referring to Section 4.3, the mean deceleration for twin engined transport aircraft leads to a deviation of less than 10% for the worst case. This means that even with this simplified deceleration estimate, the landing stall speed error should be less than 5%.

An equation for the stall speed in landing configuration was derived from landing analysis. Whereas the other parameters could be estimated from the landing conditions of transport aircraft, this expression depends on the landing field length and the braking deceleration. Based on landing performance of current airliners, an average braking deceleration was finally calculated.

Chapter 5

Stall Speed in Takeoff Configuration

The purpose of this chapter is to relate the stall speed in the takeoff configuration to its value in the landing configuration, which is obtained from the FAR part 25 landing field length analysis in the preceding chapter.

5.1 Stall Speed

For an aircraft in level flight, in a given configuration, lift equals weight and its stall speed can be calculated from:

$$W = \frac{1}{2}\rho {V_s}^2 S(x_{C_{L_{max}}} C_{L_{max}}) \tag{5.25}$$

where

- ρ : air density (sl/ft^3)
- V_s : stall speed (ft/s)
- S : wing reference area (ft^2)

- $x_{C_{L_{max}}}$: $C_{L_{max}}$ fraction for pertinent configuration
- C_{Lmax} : maximum lift coefficient

During takeoff and landing, the aircraft speed is closer to the stall speed than at any other moment of the flight. For this reason, takeoff and landing FAR requirements are described with respect to the stall speed in the corresponding configuration. Aircraft usually set their high-lift devices in different positions for takeoff and landing. Thus, two different lift coefficients are expected. Equation 5.25 is applied to these two flight phases.

5.1.1 Landing

For landing, the high-lift devices are completely extended to maximize both lift and drag ($x_{C_{L_{max}}} = 1$). Therefore, the maximum lift coefficient that an aircraft can ever achieve occurs in the landing configuration.

In [Raymer '92], the maximum landing weight is specified among the design requirements: limits are frequently between 85% and 100% of the takeoff weight. The stall speed criterion is expressed for the critical case, i.e. at the maximum landing weight. Thus,

$$W_g x_{ml} = \frac{1}{2}\rho {V_{s@l}}^2 S C_{L_{max}} \tag{5.26}$$

where

- W_g : gross weight (lb)
- x_{ml} : maximum landing over gross weight

5.1.2 Takeoff

For takeoff, the acceleration is to be maximized so that the aircraft reaches the liftoff speed at the shortest possible distance. Although the aircraft requires a high lift coefficient, the requirement for a low drag imposes a compromise. Therefore, the maximum lift coefficient is lower in takeoff than in landing configuration. Referring to [Raymer '92], the maximum lift coefficient in takeoff configuration is typically 80% of its landing value ($x_{C_{L_{max}}} = .8$). As it is shown in Section 6.2, this latter value is in full agreement with [Torenbeek] for any type of passive high-lift devices.

When the aircraft has reached its stall speed in takeoff configuration, some fuel has been burned off already. Different references present estimates of this fraction. In [Raymer '92], 97% of the gross weight remains after the start-up, taxi and takeoff, while [Roskam 1] suggests 97.52%. On the basis of those remarks, equation 5.27 is obtained.

$$W_g x_{mt} = \frac{1}{2}\rho V_{s@t}{}^2 S x_{C_{L_{max}}@t} C_{L_{max}} \tag{5.27}$$

where

- x_{mt} : maximum takeoff over gross weight
- $x_{C_{L_{max}}@t}$: $C_{L_{max}}$ fraction for takeoff configuration
- $V_{s@t}$: stall speed in takeoff configuration (ft/s)

5.2 $V_{s@t}$ Equation

The stall speed in the takeoff configuration can be related to its value in the landing configuration by taking the ratio of equations 5.26 and 5.27. Considering that the aircraft must takeoff and land from the same airport, i.e. the same altitude, equation 5.28 gives the takeoff to landing stall speed ratio.

$$\frac{x_{mt}}{x_{ml}} = (\frac{V_{s@t}}{V_{s@l}})^2 x_{C_{L_{max}}@t} \tag{5.28}$$

Solving for the stall speed in takeoff configuration, equation 5.29 is obtained:

$$V_{s@t} = V_{s@l}\sqrt{\frac{x_{mt}}{x_{ml}x_{C_{L_{max}}@t}}} \tag{5.29}$$

An approximate value for the takeoff to landing stall speed ratio can be calculated from the references. For a x_{ml} of 85%, the ratio is about 1.2 with a $x_{C_{L_{max}}@t}$ of 80% and a x_{mt} of 97.5%.

The stall speed in takeoff configuration was derived from the stall speed definition. Its expression was reduced to a function of the stall speed in landing configuration from Chapter 4 and the maximum landing over gross weight while the other parameters were kept fixed in accordance to the references cited in this chapter.

Chapter 6

Wing-Loading

First, the wing-loading at gross weight is expressed at the takeoff and the landing condition. Second, the same expression is used to determine of the maximum lift coefficient of current transport aircraft whose wing-loading is known.

6.1 $\frac{W_g}{S}$ Equation

In the preceding chapter, the stall speed criterion was expressed for landing and takeoff conditions (equations 5.26 and 5.27). These equations can now be rewritten for the wing-loading.

$$\frac{W_g}{S} = \frac{\rho {V_{s@l}}^2 C_{L_{max}}}{2x_{ml}} \tag{6.30}$$

$$\frac{W_g}{S} = \frac{\rho {V_{s@t}}^2 x_{C_{L_{max}}@t} C_{L_{max}}}{2x_{mt}} \tag{6.31}$$

During takeoff and landing, the air density is assumed to be at the sea-level standard value of .0023769 sl/ft^3. The stall speeds in landing and takeoff configurations were obtained in chapters 4 and 5, respectively. Estimates of the lift and maximum weight fractions at takeoff and landing were also presented from the references. Thus, the wing-loading reduces to a linear function of the maximum lift coefficient.

The reader may now wonder whether either equations 6.30 or 6.31 should be preferred. The choice has to take into consideration the accuracy of the parameters that are involved in these equations. Since the calculation of the stall speed in takeoff configuration is based on the stall speed in landing configuration, the former is certainly going to be less accurate than the latter. So, it is logical to use equation 6.30 for evaluating the wing-loading, since it involves or requires less calculations and approximations.

The maximum landing weight fraction and air density will be specified later so the wing-loading error will depend only on the accuracy of the landing stall speed. Unfortunately, the stall speed is squared and, thus, its error is multiplied by two. This means that special care should be taken for evaluating the stall speed in landing configuration. The quality of its evaluation affects the wing-loading accuracy.

6.2 Maximum Lift Coefficient

"Yet the estimation of maximum lift is probably the least reliable of all of the calculations used in aircraft conceptual design" [Raymer '92].

Maximum lift coefficients for current transport aircraft were among the most difficult parameters to get from the references. In [McCormick '79] and [McCormick '95], the maximum lift coefficient with partially deflected flaps at takeoff is assumed to be 1.8 and 2.1 for the B-747-100 and the B767-300ER, respectively. For a regular transport aircraft equipped with flaps and slats, [Raymer '92] gives 2.4 as a reasonable value for maximum lift coefficient. The maximum lift coefficient with flaps down is also given by [Roskam & Lan] for the B727-200, B737-200 and DC-10; they are 2.5,

Trailing edge	Leading edge	$\frac{C_{L_{max}}}{\cos\Lambda_{.25}}$@takeoff	$\frac{C_{L_{max}}}{\cos\Lambda_{.25}}$@landing
plain	-	1.4-1.6	1.7-2
single slotted	-	1.5-1.7	1.8-2.2
Fowler	-	2.0-2.2	2.5-2.9
double slotted	-	1.7-1.95	2.3-2.7
double slotted	slat	2.3-2.6	2.8-3.2
triple slotted	slat	2.4-2.7	3.2-3.5

Table 11: Typical $C_{L_{max}}$ for wing with high-lift devices [Torenbeek]

3.2 and 2.5, respectively. Finally, [Torenbeek] gives Table 11.

In Table 11, the maximum lift coefficient ratio between landing and takeoff is about 80% for any high-lift devices except double slotted flaps (73%) and triple slotted flaps with slats (76%). The average value over all the different types of high-lift devices is 78.3%. This result is in complete agreement with [Raymer '92] as it was stated in Section 5.1.2. The maximum lift coefficient is expressed by equation 6.32 using the stall speed criterion at landing:

$$C_{L_{max}} = \frac{2W_{ml}}{\rho {V_{s@l}}^2 S} \tag{6.32}$$

where W_{ml} is the maximum landing weight (lb)

In [Jane's 94-95], the maximum lift coefficient can be evaluated for several current transport aircraft. All the aircraft whose maximum landing weight, approach speed and wing reference area were indicated, are in Table 12 with their corresponding maximum lift coefficient calculated from equation 6.32. The approach speed was considered to be 130% of the stall speed at landing. The air density was taken at sea-level as usually recommended by the maximum landing weight specification. The results in Table 12 are confirmed in Table 11. For instance, the A300-600 and A310 have 28 degrees of sweepback at quarter-chord in [Jane's 94-95] , which indicates

Aircraft	W_{ml} (lb)	V_a (kts)	S (ft^2)	$C_{L_{max}}$
A300-600	304240	135	2798.6	2.973
A300-600R	308645	136	2798.6	2.972
A310-200	271170	135	2357.3	3.146
A310-200	273375	135	2357.3	3.172
A310-300	271170	135	2357.3	3.146
A310-300	273375	135	2357.3	3.172
B-757-200	198000	132	1994	2.841
B-767-200	270000	136	3050	2.386
B-767-200	272000	136	3050	2.403
B-767-200	278000	138	3050	2.386
B-767-200	278000	140	3050	2.318
B-767-200	285000	140	3050	2.376
B-767-300	300000	141	3050	2.466
B-767-300	320000	145	3050	2.487
B-777-200	445000	138	4605	2.529
B-777-200	460000	140	4605	2.540

Table 12: Calculated $C_{L_{max}}$ from W_{ml}, V_a and S [Jane's 94-95]

that the aircraft is equipped with Fowler flaps. In comparison, for the calculated maximum lift coefficient in Table 12, the use of Fowler flaps is also suggested by Table 11. Finally, for the B767-300ER with partially deflected flaps at takeoff, a maximum lift coefficient of 2.1 was assumed in [McCormick '79]. Its landing value in Table 12 from [Jane's 94-95] is about 2.5. Thus, the 'guessed' value at takeoff, 2.1, should be compares well with the suggested 80% of landing lift coefficient, 2.5, i.e., the predicted value is 2.0 for the takeoff, and the percent difference is about 5%.

The wing-loading was rederived from the stall speed definition and expressed with respect to the landing and takeoff configurations. Its expression is a function of the stall speed in the pertinent configuration in Chapters 4 and 5, the maximum lift coefficient and the maximum landing over gross weight. In order to check the validity of the model, the modified wing-loading equation was rewritten with respect to the maximum lift coefficient. From the data of current airliners, the maximum lift coefficient could then be computed and compared to the actual type of high-lift devices installed on the aircraft. The predictions of the maximum lift coefficient and its corresponding high-lift devices matched the actual devices used in existing aircraft so that the wing-loading equation was validated.

Chapter 7

Static Thrust over Gross Weight

Based on the references used in this report, either a statistical or a dynamic approach can be used to determine the static thrust over gross weight. The methods directly estimate the ratio on the basis of existing aircraft, while the latter are based on the equations of motion at a particular flight phase. Since statistical approximations are not always accurate and do not always take into account the design requirements, the analysis in this chapter is of the second type.

In [Raymer '92], the thrust-to-weight ratio is approximated in cruise and later multiplied by a constant factor to get its corresponding value at takeoff. Sizing the ratio from the cruise conditions is arguable because it is not the most critical criterion for transport aircraft. [Raymer '92] also expresses the ratio based on the equations of motion during climb. Similarly, in [Loftin] the equations of motion for the second segment of climb and the missed approach are written in terms of the thrust-to-weight ratio. Their models assume that the static thrust does not vary with speed and that the aircraft initiates its climb and approach at its takeoff weight. In their analyses, approximate or statistical values of the aerodynamic coefficients are used. For transport aircraft, the discrepancies between the results of the previous prediction methods and the actual values of the static thrust over gross weight appear to be unacceptably too large. Therefore, the prediction method needs to be improved for our design synthesis problem.

7.1 Equations of Motion

Starting with the equations of motion in the vertical plane, equations 7.33 and 7.34 govern the aircraft dynamics, tangent and normal to the flight path, respectively.

$$\frac{W}{g}a_t = Tcos\phi - D - Wsin\gamma \tag{7.33}$$

$$\frac{W}{g}a_c = Tsin\phi + L - Wcos\gamma \tag{7.34}$$

where

- T : thrust (lb)
- D : drag (lb)
- a_t : tangential acceleration (ft/s^2)
- ϕ : thrust-flight path angle
- γ : climb-descent angle

Equations 7.33 and 7.34 can be simplified. The thrust is considered to be aligned with the flight path consistent with the references. The normal climb, cruise and descent flight segments are also approximated by straight lines. It is further assumed that the flight segments are connected by smooth transition lines. The centripetal acceleration is assumed to be negligible. Then, the equations of motion are reduced to equations 7.35 and 7.36.

$$\frac{W}{g}a_t = T - D - Wsin\gamma \tag{7.35}$$

$$L = Wcos\gamma \tag{7.36}$$

In order to maintain the equilibrium described by equation 7.36, it is implicitly expected that the pilot controls the lift coefficient to compensate for the change in weight, altitude and speed due to the acceleration along a straight flight path. Equation 7.35 is divided by the weight to get the non-dimensional equation 7.37.

$$\frac{a_t}{g} + sin\gamma = \frac{T}{W} - \frac{D}{L}cos\gamma \tag{7.37}$$

Except for level flight, the acceleration within the load factor can be further expanded as the product of the speed gradient, $\frac{dV}{dh}$, and the rate of climb: $\frac{dh}{dt} = V sin\gamma$.

$$\frac{dV}{dt} = \frac{dV}{dh}\frac{dh}{dt} = \frac{dV}{dh}V sin\gamma \tag{7.38}$$

Equation 7.39 defines the acceleration factor for climbs and descents.

$$\frac{V}{g}\frac{dV}{dh} = c_a M^2 \tag{7.39}$$

where

- c_a : acceleration constant for climb
- M : Mach number

For a climb in the troposphere either at constant equivalent airspeed or at constant Mach number, [Roskam & Lan] demonstrates that this factor may be expressed as the product of a constant c_a and the square of the Mach number. [Torenbeek] mentions also the same relationships in the troposphere and gives the corresponding constant for identical climb schedules in the stratosphere. For a climb at constant equivalent airspeed, the constant c_a is .5668 in the troposphere and .7 in the stratosphere. For a climb at constant Mach number, the constant c_a is -.133 in the troposphere and 0 in the stratosphere. [Roskam & Lan] also indicates that a climb at constant calibrated airspeed allows the best rate-of-climb to be maintained. Pilots usually follow this climb schedule until the aircraft reaches the initial cruise Mach number at which time a climb at constant Mach number is used. By combining equations 7.38 and 7.39, the load factor is finally reduced to equation 7.40.

$$\frac{a}{g} = c_a M^2 sin\gamma \tag{7.40}$$

The load factor from equation 7.40 is then substituted into equation 7.37, which is then rewritten for the thrust-to-weight ratio to get equation 7.41.

$$\frac{T}{W} = cos\gamma \frac{D}{L} + sin\gamma(1 + c_a M^2) \tag{7.41}$$

7.2 FAR Part 25 Climb Requirements

Referring to equation 7.41, it is expected that critical flight phases for transport aircraft are during takeoff and landing, when the aircraft speed is close to the stall speed. Accordingly, FAR part 25 states emergency climb requirements associated with takeoff and landing. If the static thrust to gross weight ratio is sized for the more restrictive emergency situation, it is expected that this static thrust over gross weight will satisfy all other constraints. This is verified later and the results are presented in Table 14. The present approach provides a more realistic results than a sizing based on the cruise conditions as in [Raymer '92].

7.2.1 First-Segment of Climb

The first-segment of climb, also called takeoff climb potential, starts just after liftoff and ends as the landing gear is fully retracted. Although this flight phase occurs close to the ground, ground effect is ignored. Also, during this segment, the high-lift devices are in takeoff configuration. The aircraft is assumed to fly at liftoff speed with maximum takeoff thrust at which time one engine becomes inoperative. Liftoff speed should be greater than 110% of stall speed in the takeoff configuration.

The aircraft should then be able to maintain a minimum climb gradient which varies with the number of engines: greater than 0% for twin engines, .3% for three engines and .5% for four engines.

7.2.2 Second-Segment of Climb

The second-segment of climb begins when the landing gear is retracted and ends at an altitude of 400 feet. During this segment, the high-lift devices are still in takeoff position. The aircraft is assumed to fly at takeoff safety speed (above 35 ft) and maximum takeoff thrust with one engine inoperative. Takeoff safety speed should be greater than 120% of stall speed in the takeoff configuration. The aircraft should then be able to maintain a minimum climb gradient: 2.4% for twin engines, 2.7% for three engines and 3.0% for four engines.

7.2.3 Third-Segment of Climb

The third-segment of climb, also called final takeoff, occurs between an altitude of 400 feet and 1500 feet. During this segment, the high-lift devices are now retracted and the thrust is reduced to the maximum continuous rating while the aircraft continues to accelerate. The aircraft flies at a speed greater than 125% of the stall speed in clean configuration and maximum continuous thrust with one engine out. It should then be able to maintain a minimum climb gradient: 1.2% for twin engines, 1.4% for three engines and 1.5% for four engines.

7.2.4 Go-Around in Approach Configuration

The missed approach, or approach climb potential, is assumed to take place close to the ground but still outside ground effect. During this segment, the landing gear is still retracted. The high-lift devices are set in an approach position such that the corresponding stall speed is at least 110% of its value in the landing configuration. The aircraft flies at maximum takeoff thrust with a speed greater than 150% of the stall speed in the approach configuration with one engine inoperative. The aircraft should then be able to maintain a minimum climb gradient: 2.1% for twin engines, 2.4% for three engines and 2.7% for four engines.

7.2.5 Go-Around in Landing Configuration

The missed landing, or landing climb potential, is also assumed to happen close to the ground but, again, still outside ground effect. Now, the landing gear is extended and the high-lift devices are set in landing configuration. The aircraft flies with a speed greater than 130% of stall speed in the landing configuration and all its engines are at maximum takeoff thrust. More precisely, it should be the engine thrust available 8 seconds after opening the throttle to takeoff rating. The aircraft should then be able to maintain a minimum climb gradient of 3.2%. According to [Torenbeek], this requirement is usually not critical because modern turbine engines have a fast response, and the thrust reaches its maximum value without too much lag.

7.3 Full Throttle Thrust at Sea Level

Since the critical flight phase occurs close to the ground, i.e. during emergency takeoff and landing, the altitude is approximately at the airport altitude. The airport is assumed to be at sea level. Except for the third-segment of climb, FAR part 25 allows the pilot to use full throttle in order to recover the aircraft from an emergency situation. Thus, an expression for the full throttle thrust at sea level is developed next.

The idea is based on the takeoff analysis in [McCormick '79] and [McCormick '95]. The takeoff thrust is expressed as a second order polynomial of the aircraft speed. Such an expression could also be used for emergency operations following the takeoff and preceding the landing. It is expected to be more accurate than the static thrust approach used by [Raymer '92] and [Loftin].

Equation 7.42 is taken from [McCormick '79]. It is the takeoff thrust of a Pratt & Whitney JT9D-7A (by-pass ratio of 5.1) at sea level, different versions of which equip the B-747. Equation 7.44 from [McCormick '95] gives the takeoff thrust of a Pratt & Whitney PW4056 (by-pass ratio of 4.9) at sea level. This engine is installed on current versions of B747-400, B-767-200/300, MD-11, A300-600 and A310-300. In [Raymer '92], the full throttle thrust at sea level is graphed for a high-bypass ratio turbofan (by-pass ratio of 8.0) which is said to be representative of modern turbofan engines. Equation 7.43 was obtained from this graph by fitting a quadratic equation.

$$T = 46100 - 46.7V + .0467V^2 \tag{7.42}$$

$$T = 46100(1 - 1.013(10)^{-3}V + 1.013(10)^{-6}V^2)$$

$$T = 49553 - 49.9V + .0330V^2 \tag{7.43}$$

$$T = 49553(1 - 1.008(10)^{-3}V + .666(10)^{-6}V^2)$$

$$T = 55600 - 46.0V + .0357V^2 \tag{7.44}$$

$$T = 55600(1 - .827(10)^{-3}V + .642(10)^{-6}V^2)$$

where V : speed (ft/s)

When equations 7.42, 7.43 and 7.44 are written with respect to the maximum static thrust, the right-hand side terms are normalized quadratic polynomials whose coefficients do not depend appreciably on the engine, especially for turbofan engines with a by-pass ratio in the range of 4.9 to 8. For such turbofan engines, the full throttle thrust at sea level can be represented as the product of the maximum static thrust and a calibrated quadratic polynomial with respect to the speed. This model assumes that these three normalized quadratic equations are very close in the interval considered for evaluating the thrust-to-weight ratio. The maximum approach speed of transport aircraft is about 150 knots. Based on Section 7.2, it can be shown that the maximum aircraft speed should never exceed 175 knots (Mach .26) during any FAR part 25 emergency climb performance. Within the speed interval of 0 to 175 knots, the error between the three normalized quadratic equations is less than 3%. Then a normalized quadratic equation that best represents these three equations, can be obtained (see Appendix B). The three curves are sampled with increasing speeds so that an average normalized thrust could be calculated for any given speed. Equation 7.45 is obtained by approximating the average normalized thrusts with respect to the aircraft speed. The maximum difference between equation 7.45 and the average normalized thrust was less than .01%. To conclude, equation 7.45 approximates equations 7.42, 7.43 and 7.44 with less than 3% error for a given static thrust,

$$T = T_{s/e}(1 + c_v V + c_{v^2} V^2) \tag{7.45}$$

where

- $T_{s/e}$: static thrust per engine (lb)
- $c_v = -.949(10)^{-3} \frac{lb}{ft/s^2}$
- $c_{v^2} = .773(10)^{-6} \frac{lb}{(ft/s^2)^2}$

$$T_s = n_T T_{s/e} \tag{7.46}$$

where

- T_s : static thrut (lb)
- n_T : number of engines

The total thrust for an aircraft is defined by equation 7.46. Finally, referring to [McCormick '79] and [McCormick '95], the maximum continuous static thrust at sea level is 39650 lb for the JT9D-7A and 48000 lb for the PW4056. Thus, the ratio of maximum continuous thrust to maximum takeoff thrust ratio can be calculated at sea level, i.e., 87.14% and 86.33%, respectively. These are in full agreement with [Torenbeek].

7.4 $\frac{T_s}{W_g}$ Equation

The thrust-to-weight ratio from equation 7.41 can now be expressed with respect to the total static thrust to gross weight ratio according to equation 7.45. Equation 7.47 is also a function of the number of operating engines, instantaneous weight and thrust of the aircraft.

$$\frac{T}{W} = \frac{n}{n_T} \frac{x_t(1 + c_v V + c_{v^2} V^2)}{x_w} \frac{T_s}{W_g} \tag{7.47}$$

where

- n : number of operating engines
- x_w : instantaneous weight fraction
- x_t : instantaneous thrust fraction

The thrust-to-weight ratio expression from equation 7.41 can be substituted into the equation 7.47 and rewritten for the static thrust to gross weight ratio. The Mach number from equation 7.41 can be converted to speed using the speed of sound. Since the altitude is given, the speed of sound is known.

$$\frac{T_s}{W_g} = \frac{n_T}{n} \frac{x_w}{x_t(1 + c_v V + c_{v^2} V^2)} \left(\frac{C_D}{C_L} + sin\gamma(1 + c_a(\frac{V}{a})^2)\right) \tag{7.48}$$

where a is the speed of sound (ft/s)

The speed appearing in equation 7.48 can be expressed as a fraction of the stall speed in the pertinent condition according to equation 7.49.

$$V = x_v V_s \tag{7.49}$$

where x_v is the instantaneous stall speed fraction

The lift coefficient is determined on the basis of the stall speed definition given in equation 5.25. As stated in equation 7.50, the aircraft lift at a given speed must equal its expression at stall speed. After simplification, the lift equation 7.51 is obtained.

$$\frac{1}{2}\rho(x_v V_s)^2 S C_L = W = \frac{1}{2}\rho {V_s}^2 S(x_{C_{L_{max}}} C_{L_{max}}) \tag{7.50}$$

$$C_L = \frac{x_{C_{L_{max}}} C_{L_{max}}}{{x_v}^2} \tag{7.51}$$

where C_L is the lift coefficient

Finally, the drag is composed of the parasite and induced drag components. The parasite drag coefficient consists of the clean configuration parasite drag coefficient plus the contribution of the extended high-lift devices and landing gear. The parasite drag coefficient in clean configuration is independent of the flight phase, but the increment in parasite drag coefficient will vary according to the flight phase: the landing gear and the high lift-devices may be completely extended or retracted. Sometimes, slats and flaps may even be partially deployed. The induced drag component is generated by the lift and is also a function of the wing shape. The wing aspect ratio and the Oswald efficiency factor which appear in the induced drag term, take into account this wing shape dependence. Whereas the wing aspect ratio is fixed, the Oswald efficiency factor varies with the high-lift devices configuration. Therefore, the

Oswald efficiency factor for a given configuration was defined as a fraction x_e of its value for clean configuration.

$$C_D = C_{D_o} + \Delta C_{D_o} + \frac{{C_L}^2}{\pi A(x_e e)} \tag{7.52}$$

where

- C_D : drag coefficient
- C_{D_o} : clean configuration parasite drag coefficient
- ΔC_{D_o} : increment of parasite drag coefficient
- A : wing aspect ratio
- x_e : Oswald efficiency fraction
- e : clean configuration Oswald efficiency factor

The parasite drag coefficients and the Oswald efficiency factor are flight phase dependent; they are discussed in Section 7.5 and 7.6, respectively. They should be estimated for the different FAR part 25 climb requirements which were described in Section 7.2.

7.5 Parasite Drag Coefficient

The references used suggest two methods for estimating the parasite drag coefficient. The equivalent skin-friction, or flat plate, method is presented in [McCormick '79], [McCormick '95], [Raymer '92] and [Roskam & Lan]. The component buildup method is described in [Raymer '92], [Roskam & Lan] and [Torenbeek].

However, both methods are difficult to apply at the conceptual design level. On the one hand, the equivalent skin-friction method requires the total aircraft wetted area, which has not been determined at this stage of the design, e.g., the wing and the tail surface areas are unknown. On the other hand, the component buildup method is based on the knowledge of an even more detailed description of the aircraft. The principle of adding the components effects also leads to an accumulation of errors since some terms are only statistical estimates at best and others are even greater uncertain approximations such as for the landing gear. According to [Raymer '92], numerous estimation methods overestimate the actual value of the parasite drag coefficient.

Using the equivalent skin friction method, [McCormick '79] and [McCormick '95] calculated the parasite drag coefficient of two transport aircraft at takeoff and climb configurations: .036 and .018 for a B-747 ; .041 and .014 for a B-767.

[Raymer '92] gives an initial estimate of .015 as the parasite drag coefficient of a jet aircraft in cruise. Due to the high-lift devices and the landing gear, the parasite drag coefficient needs to be corrected for takeoff, climb, glide and landing configurations. For takeoff flaps and slats settings, an increment of .02 is added to the clean parasite drag coefficient. For high-lift devices settings at landing, an increment of .07 is suggested. Moreover, there is an additional increment of about .02 in parasite drag coefficient for the extended landing gear. Finally, the drag coefficient increase from a stopped turbofan engine is neglected for the initial analysis.

In order to calculate the parasite drag coefficients in clean configuration in Table 13, [Roskam & Lan] adopted an Oswald efficiency factor of .85. [Torenbeek] suggests that the typical parasite drag coefficients range between .014 and .020 for a high-subsonic jet aircraft in cruise configuration. At takeoff safety speed, the parasite drag increment due to extended slats is .018 and .005 with slats retracted.

aircraft	C_{D_o}
B-707-320B	.0131
DC8-63	.0156
B-727-200	.0173
B-737-200	.019
DC9-30	.0196
B747-100	.0148
DC10-30	.0162
L1011-1	.0161
Airbus B2	.0171

Table 13: Estimated clean configuration C_{D_0} [Roskam & Lan]

7.6 Oswald Efficiency Factor

For calculating the Oswald efficiency factor, a statistical relation that is valid for sweep angles greater than 30 degrees is given in [Raymer '92], but it yields very low values that seem unrealistic. According to the same author, other estimation methods overestimate the actual values. Finally, the leading edge suction method is also presented in [Raymer '92], but it requires a more detailed description of the wing geometry than is available at this stage of the design. [Raymer '92] states that, as a first approximation, an Oswald efficiency factor of .8 in cruise for any aircraft except fighters. Also, this value is said to vary between .7 and .85. Some corrections for high-lift devices and landing gear effects are also given for takeoff, climb, glide and landing configurations. The takeoff configuration decreases the Oswald efficiency factor by about 5%. The Oswald efficiency factor is also decreased by about 10% for landing flaps and slats settings. [Loftin] uses an Oswald efficiency factor of .7 for transport aircraft during takeoff, second segment of climb, missed approach and landing. For takeoff and climb of a B-747 and a B-767, [McCormick '79] and [McCormick '95] uses an Oswald efficiency factor of .7 for the aircraft. The same author also states, without specifying the configuration, that typical values for a low-wing and a high-wing aircraft are about .6 and .8, respectively. [Roskam & Lan] suggests an Oswald efficiency factor of .75, but a value of .85 is chosen in order to calculate the cruise configuration parasite drag of transport aircraft. According to [Torenbeek], a high-subsonic jet aircraft in cruise configuration has an Oswald efficiency factor between .75 and .85. It also states that as the sweep angle is increased, the Oswald efficiency factor decreases. At takeoff safety speed, the Oswald efficiency factor is .7 with extended slats and .61 if the slats are retracted. For low thrust-to-weight ratio aircraft, the drag due to a failed engine decreases the Oswald efficiency factor by 4% for wing-mounted engines and 2% for fuselage-mounted engines.

7.7 $\frac{T_s}{W_g}$ Criteria

The static thrust to gross weight ratio can be calculated using equations 7.48, 7.49, 7.51, 7.52. The aircraft parameters appearing in these equations need to be chosen for the conditions stated in the FAR part 25 climb requirements that are described in Section 7.2. These requirements produce different design criteria (first, second and third segments of climb, missed approach and landing) and the results are shown for several current transport aircraft in Table 14.

Equation 7.48 : $\frac{T_s}{W_g} = \frac{n_T}{n}\frac{x_w}{x_t(1+c_vV+c_{v2}V^2)}(\frac{C_D}{C_L} + sin\gamma(1+c_a(\frac{V}{a})^2))$ The gross weight fractions x_w for the first, second and third segments of climb is taken to be identical without loss of accuracy. [Raymer '92] and [Roskam 1] suggest 97% and 97.52%, respectively. The former value was arbitrarily adopted for the calculation. The maximum value of the gross weight fraction for approach and landing is the maximum landing weight fraction x_{ml}. It was calculated from the maximum takeoff and landing weights in [Jane's 94-95] for the different aircraft. Except for the third segment of climb, the maximum value of the takeoff thrust was used, i.e., $x_t = 1$. A ratio of the maximum continuous to takeoff thrust of about 87% was estimated in Section

7.3. This value is used for the third segment of climb, i.e., $x_t = .87$. Whereas the aircraft undergoes an acceleration for normal climb and descent, it was assumed that the acceleration factor c_a vanishes during the FAR part 25 climb requirements since the FAR requirements place no limit on the acceleration.

Equation 7.49 : $V = x_v V_s$ All the speeds are always assumed to be the minimum FAR part 25 values for the climb requirements.

Equation 7.51 : $C_L = \frac{x_{C_{Lmax}} C_{Lmax}}{x_v{}^2}$ Due to the takeoff flaps and slats settings, the maximum lift coefficient for the first and second segment of climb is assumed to be 80% of the maximum lift coefficient. The lift coefficients at missed landing and missed approach are calculated with the respective values x_v and $x_{C_{L_{max}}} = 1$. Although $x_{C_{L_{max}}} = 1$ is a common approximation for the missed landing configuration, it is not the case for the missed approach. However, $x_{C_{L_{max}}} = 1$ can still be used for the missed approach since the FAR regulations specify the missed approach criterion with respect to the landing configuration, i.e., $x_{C_{L_{max}}} = 1$, provided the value of x_v is selected with respect to the corresponding stall speed, see Section 7.2.5. The lift fraction for the third segment of climb is more difficult to handle because the flaps and slats are fully retracted. The flaps and slats settings at takeoff and landing were found in [Torenbeek] for different types of high-lift devices, and the lift coefficient is estimated to decrease by 20% between landing and takeoff settings. Assuming the lift coefficient increment is proportional to the deflection angle of the flaps, the maximum lift coefficient with fully retracted slats and flaps are approximately 60 to 70% of the maximum lift coefficient in landing configuration. An average value of 65% was taken for the calculation. This latter fraction corresponds to the third segment of climb speed equal to 110% of the stall speed in clean configuration, which is used frequently as a lower bound for the clean configuration speed.

Equation 7.52 : $C_D = C_{D_o} + \Delta C_{D_o} + \frac{{C_L}^2}{\pi A(x_e e)}$ Referring to Section 7.5, an average value of .016 is assumed for the clean configuration parasite drag coefficient. The increment of parasite drag coefficient varies with high-lift devices settings and the landing gear position: .04, .02 and 0 for the first, second and third segments of climb, respectively; .09 for the missed landing; and .045 for the missed approach. Those values come directly from Section 7.5, except for the last one. It was assumed that the parasite drag increment in the approach and the landing configurations are about the same $(\frac{1}{2}\rho {V_a}^2 S(\Delta C_{D_0})_a = \frac{1}{2}\rho {V_l}^2 S(\Delta C_{D_0})_l)$. Finally, from Section 7.6 an Oswald efficiency factor of .7 is selected for all the climb criteria.

In Table 14, the different criteria are calculated for the twin turbofan transport aircraft whose approach speeds are available in [Jane's 94-95], see Appendix C. The Loftin criteria for second segment of climb and missed approach are also calculated using the lift to drag formula previously described rather than the statistical method in [Loftin], see Appendix C. The required static thrust over gross weight should be greater or equal to the maximum value from the different criteria. It is observed that the new method produces results closer to actual values than Loftin criteria. [Loftin] correctly observed that the critical thrust-to-gross weight ratio would is more likely to occur either for the second segment of climb or the missed approach. Most of the time, the second segment of climb seems to be the critical sizing criterion. Therefore, the other criteria are not considered.

In Table 14, $\frac{T_s}{W_g}$ is calculated for:

- 1sc : first segment of climb
- 2sc : second segment of climb
- 2scL : second segment of climb (Loftin)
- 3sc : third segment of climb

aircraft	1sc	2sc	2scL	3sc	ma	maL	ml	max	actual
A300-600	.340	.340	.283	.272	.299	.282	.199	.340	.338
A300-600R	.341	.341	.283	.273	.294	.282	.195	.341	.327
A310-200	.317	.320	.268	.255	.293	.267	.195	.320	.319
A310-200	.318	.321	.269	.256	.293	.267	.195	.321	.319
A310-300	.318	.321	.268	.256	.277	.267	.184	.321	.323
A310-300	.320	.323	.269	.258	.277	.267	.185	.323	.317
B757-200	.326	.327	.275	.261	.304	.280	.201	.327	.325
B757-200	.329	.330	.275	.263	.280	.280	.185	.330	.333
B767-200	.295	.298	.250	.234	.320	.281	.205	.320	.320
B767-200	.296	.300	.251	.236	.307	.281	.197	.307	.304
B767-200	.298	.302	.250	.237	.288	.281	.184	.302	.290
B767-200	.295	.299	.247	.235	.290	.282	.184	.299	.304
B767-200	.297	.302	.247	.237	.265	.282	.168	.302	.299
B767-300	.302	.305	.254	.241	.311	.280	.200	.311	.304
B767-300	.302	.306	.254	.241	.305	.280	.196	.306	.299
B767-300	.305	.308	.254	.243	.277	.280	.178	.308	.299
B767-300	.306	.310	.254	.245	.287	.280	.185	.310	.300
B777-200	.286	.292	.244	.229	.302	.271	.194	.302	.304
B777-200	.286	.292	.244	.230	.297	.271	.191	.297	.299
B777-200	.287	.293	.244	.230	.286	.271	.184	.293	.288
B777-200	.290	.296	.244	.232	.273	.271	.176	.296	.290

Table 14: Calculated $\frac{T_s}{W_g}$ from V_a, $\frac{W_g}{S}$, x_{ml} and A [Jane's 94-95]

- ma : missed approach
- maL : missed approach (Loftin)
- ml : missed landing
- max : maximum of 1sc, 2sc, 3sc, ma, ml
- actual : Jane's 94-95

The estimate of the constant parameters in equations 7.48, 7.49, 7.51, 7.52 should be refined in order to improve the sizing. Although the clean parasite drag coefficient of transport aircraft may vary between .013 and .02, such variations of clean configuration parasite drag coefficient have little influence on the overall drag coefficient in the FAR climb requirement calculations, see equation 7.52. For the worst case, i.e., a low value of the maximum lift coefficient (2.2) for a transport aircraft and a high wing aspect ratio (8.8), an error of 4 counts in the clean configuration parasite drag coefficient generates a propagated error less than 3% of the total drag coefficient calculated with equation 7.52. The choice of a clean parasite drag coefficient independent of the aircraft design is therefore justified for the purpose of obtaining an estimate of the static thrust over gross weight. On the other hand, the estimation of the increments of parasite drag coefficient and the Oswald efficiency factor need to be improved: they are the major source of errors.

In this chapter, an estimate of the static thrust over gross weight was obtained from the FAR part 25 climb requirements, i.e., the first, second and third segments of climb and the missed approach and landing. The model takes into account the thrust variation with speed, the actual weight of the aircraft and its acceleration. The stall speeds in takeoff and landing configurations from Chapters 5 and 6, the wing aspect ratio and the maximum lift coefficient are required in order to estimate the static thrust over gross weight. This new model was tested on current transport aircraft from [Jane's 94-95] and lead to closer results than the Loftin's method.

Chapter 8

Cruise Weight Fraction

Whereas most of the weight fractions for the flight segments can be considered constant for the conceptual design of a jet transport aircraft, the cruise weight fraction may drastically change with the design cruising conditions. For jet aircraft, the cruise weight fraction is a function of the Thrust Specific Fuel Consumption (TSFC). The TSFC is defined as the fuel weight that is consumed per second for every pound of thrust. Thus, the rate of change of aircraft weight during the flight can be written as equation 8.53.

$$dW = -C_{TSFC} T dt \tag{8.53}$$

where C_{TSFC} : thrust specific fuel consumption (1/s)

If the right hand-side of equation 8.53 is multiplied by $\frac{V}{\frac{dR}{dt}}(=1)$, equation 8.54 is obtained, where the speed and distance are measured in knots and nautical miles, respectively.

$$dW = -\frac{C_{TSFC} T}{V} \frac{dR}{dt} dt = -\frac{C_{TSFC} T}{V} dR \tag{8.54}$$

During the cruise, thrust equals drag and lift equals weight. The thrust may then be written as a function of the weight:

$$T = D = \frac{C_D}{C_L} L = \frac{C_D}{C_L} W \tag{8.55}$$

By substituting the thrust from equation 8.55 into equation 8.54, equation 8.56 is obtained. It represents the fuel weight consumed per infinitesimal displacement along the cruise path.

$$dW = -\frac{C_{TSFC}}{V} \frac{C_D}{C_L} W dR \tag{8.56}$$

In general, the parabolic drag polar approximation applies to moderate to high wing aspect ratio aicraft flying at low Mach number. In contrast, transport aircraft have a high wing aspect ratio and are usually designed to fly at high cruise Mach number, but below the drag divergence Mach number. However, in [Raymer '92], [Roskam & Lan] and [Torenbeek], the parabolic drag relation is extensively used for the cruise analysis.

Since equation 8.56 is later integrated from the begining to the end of the cruise, it is important to understand the behavior of the right hand-side of equation 8.56 with respect to the range. The range is always a design requirement. In [Torenbeek], ATA '67 states that for transport aircraft reserve fuel should allow for a 200 nautical miles deviation at cruising speed to an alternate airport. The corresponding altitude is optional for domestic operations, i.e., for a range below 3000 nautical miles, and is the cruising altitude for best range for international operations, i.e., for a range over 3000 nautical miles. However, the altitude for which the 200 nautical miles detour equals the climb plus descent range should not be exceeded. A convenient approximation to take into consideration this reserve fuel would be to add the 200 nautical miles to the design range.

8.1 TSFC in Cruise

[Raymer '92] states that the TSFC is essentially independent of the speed for subsonic aircraft. [Roskam & Lan] and [Torenbeek] agree that the engine specific fuel consumption is constant. However, TSFC actually varies with altitude and Mach number to a small degree. Referring to turbofan engines characteristics in [McCormick '79] and [McCormick '95], the TSFC can be considered constant within an error of 3% for cruising altitudes between 25000 and 45000 feet and a cruising Mach number around .8. The TSFC at maximum cruising thrust varies between .68 and .7 for a JT9D-7A and between .58 and .6 for a PW4056. [Raymer '92] suggests a cruising TSFC of .8 for low-bypass ratio turbofan engines and .5 for high-bypass ratio turbofan engines. In [Jane's 94-95] the CFM-56 series has a cruising TSFC between .567 and .661 depending on the engine model; the corresponding TSFC is .575 for a Rolls-Royce V2500. On this basis, the TSFC is assumed to be constant for the entire range. For the conceptual study, the designer is advised to select an average TSFC for current technology engines.

8.2 Cruise Speed

Given a constant TSFC, the speed that minimizes the right hand-side of equation 8.56 can be determined. On the basis of the parabolic drag polar approximation and the lift-weight equilibrium in cruise, the drag force can be decomposed into parasite and induced drags as described by equations 8.57 and 8.58, respectively.

$$D_{parasite} = \frac{1}{2}\rho V^2 S C_{D_o} \tag{8.57}$$

$$D_{induced} = \frac{1}{2}\rho V^2 S \frac{{C_L}^2}{\pi A e} = \frac{L^2}{\frac{1}{2}\rho V^2 S \pi A e} = \frac{W^2}{\frac{1}{2}\rho V^2 S \pi A e} \tag{8.58}$$

Substituting into equation 8.56,

$$\frac{1}{V}\frac{C_D}{C_L} = \frac{1}{V}\frac{D}{L} = \frac{C_p V + C_\imath V^{-3}}{L} \tag{8.59}$$

where

- $C_p = \frac{1}{2}\rho S C_{D_o}$
- $C_\imath = \frac{2W^2}{\rho S \pi A e}$

Taking the derivative of equation 8.59 with respect to the speed, equation 8.60 is obtained. Its real positive root corresponds to the optimal speed, which minimizes equation 8.59.

$$\frac{d(\frac{1}{V}\frac{C_D}{C_L})}{dV} = \frac{C_p - 3\frac{C_\imath}{V^4}}{L} = 0 \tag{8.60}$$

Equation 8.61 gives the optimal cruising speed with respect to the instantaneous aircraft weight and the local air density while the remaining parameters in the expression are determined from the given aircraft. Since the weight decreases during the cruise, the flight altitude and its variation during the cruise basically determines the optimal speed. But, as the optimal cruising speed increases, the fuel consumption decreases (see equation 8.56). Thus, referring to equation 8.61, the range increases with cruising altitude. However, the cruising altitude is limited by the thrust required to fly at higher speed and less dense air.

$$V_{oc} = \sqrt[4]{\frac{12(W/S)^2}{\rho^2 C_{D_o} \pi A e}} \tag{8.61}$$

where V_{oc} is the optimal cruising speed (ft/s)

If equation 8.61 is written for sea level conditions, equation 8.62 is obtained. By rewriting equation 8.61 with respect to the equivalent speed, it shows that, at a given moment, the optimal calibrated airspeed is a constant independent of the altitude.

However, this calibrated airspeed, given in equation 8.62, decreases with the aircraft weight.

$$V_{oc@sl} = \sqrt[4]{\frac{12(W/S)^2}{{\rho_{sl}}^2 C_{D_o} \pi A e}} \tag{8.62}$$

$$V_{oc} = \frac{V_{oc@sl}}{\sqrt{\frac{\rho_{oc}}{\rho_{sl}}}} \tag{8.63}$$

where

- $V_{oc@sl}$: optimal cruising equivalent speed (ft/s)
- ρ_{sl} : air density at sea level (sl/ft^3)
- ρ_{oc} : air density at optimal cruising altitude (sl/ft^3)

Equation 8.63 may be written as a function of Mach number if it is divided by the speed of sound at cruising altitude and the right hand-side of the equation is expressed with respect to the speed of sound at sea level. Then, using the perfect gas law, the temperature ratio (from the conversion of speeds of sound) and the density ratio, equation 8.64 can be written in terms of the pressure ratio.

$$M_{oc} = M_{oc@sl} \sqrt{\frac{\rho_{sl}}{\rho_{oc}}} \sqrt{\frac{T_{sl}}{T_{oc}}} = M_{oc@sl} \sqrt{\frac{p_{sl}}{p_{oc}}} \tag{8.64}$$

where

- M_{oc} : optimal cruising Mach number
- $M_{oc@sl}$: equivalent optimal cruising Mach number at sea level
- T : air temperature (degrees)
- p : air pressure (lb/in^2)

and the subscripts *sl* and *oc* denote sea level and optimal cruise conditions, respectively.

The temperature and pressure ratios with respect to sea level ($\frac{p}{p_{sl}}$ and $\frac{T}{T_{sl}}$) vary with altitude and are given in equation 8.65 where the altitude in the troposphere is measured in feet. Above 36,098 feet, the temperature is constant in the stratosphere and the pressure variation is given by equation 8.66. Both equations come from the U.S. standard atmosphere.

$$\frac{T}{T_{sl}} = 1 - 6.875(10)^{-6}h = (\frac{p}{p_{sl}})^{1/5.2561} \tag{8.65}$$

$$.2234e^{-\frac{h-36089}{20806.7}} = \frac{p}{p_{sl}} \tag{8.66}$$

where h is the altitude (ft)

By substituting the pressure ratio from equation 8.65 and 8.66 into equation 8.64, an expression relating the optimal cruising altitude to the Mach number at every instant of the cruise is obtained. If the previous relation is rewritten with respect to the cruising altitude, the optimum altitude for cruise is calculated for a required cruising Mach number as shown in equation 8.67 in the troposphere and 8.68 in the stratosphere.

$$h_{oc} = \frac{1}{6.875(10)^{-6}}(1 - (\frac{M_{oc@sl}}{M_c})^{2/5.2561}) \tag{8.67}$$

$$h_{oc} = 36089 + 20806.7ln(.2234(\frac{M_c}{M_{oc@sl}})^2) \tag{8.68}$$

where

- h_{oc} : optimal cruising altitude (ft)
- M_c : cruising Mach number

Equations 8.67 and 8.68 are valid only if the optimal cruising altitude is within the troposphere or the stratosphere, respectively. This expression was applied to different existing transport aircraft whose required data were available in [Jane's 94-95]. For jet transport aircraft that usually fly at a cruising Mach number of around .8, the best cruising altitude was always in the troposphere and also below the actual initial cruising altitude stated in [Jane's 94-95]. Indeed, for transport aircraft cruising in the troposphere, the cruising speed calculated from the cruising Mach number and altitude given in [Jane's 94-95] was about 85% of the optimal value calculated with equation 8.61. This means that transport aircraft do not generally cruise at optimal conditions. Air traffic controllers usually assign a cruising altitude. Certainly, pilots try to approach the optimal cruising speed given by equations 8.67 and 8.68 for a prescribed altitude as long as it does not exceed the drag divergence Mach number. If they have the opportunity to choose their flight altitude, as it is usually the case for early morning and night flights, they should fly at the optimal cruising altitude given by equation 8.67 and the corresponding Mach number.

Since the cruise weight fraction needs to be found for the actual flight condition (usually not the optimal condition), equations 8.69 and 8.70, given for altitudes below and above 36089 ft respectively, determine the speed in cruise from the Mach number and altitude suggested in [Jane's 94-95].

$$V_c = M_c a_{sl}\sqrt{\frac{T_c}{T_{sl}}} = 1116.4 M_c \sqrt{1 - 6.875(10)^{-6} h_c} \tag{8.69}$$

$$V_c = 968.1 M_c \tag{8.70}$$

Here, the subscript c denotes the actual cruise condition.

8.3 Drag to Lift Ratio in Cruise

According to [McCormick '79] and [McCormick '95], transport aircraft cruise at constant lift coefficient. [Roskam & Lan] also states that jet aircraft fly at a constant angle of attack in cruise. [Raymer '92] and [Torenbeek] both use the optimal cruise lift coefficient for estimating the cruise weight fraction. Thus, the same assumption is made in the present analysis.

The cruising speed is expressed as a function of the lift coefficient using the lift-weight balance. Equation 8.71 gives the speed at any instant of the cruise.

$$V = \sqrt{\frac{2W}{\rho S C_L}} \tag{8.71}$$

The right-hand side of equation 8.56 is minimized with respect to the lift coefficient. Using equation 8.71 and the parabolic drag polar approximation, equation 8.72 is obtained.

$$\frac{1}{V}\frac{C_D}{C_L} = \sqrt{\frac{\rho S C_L}{2W}}\frac{C_{D_o} + \frac{C_L{}^2}{\pi Ae}}{C_L} = \sqrt{\frac{\rho S}{2W}}\left(\frac{C_{D_o}}{\sqrt{C_L}} + \frac{\sqrt{C_L{}^3}}{\pi Ae}\right) \tag{8.72}$$

It can be easily observed that the positive root of equation 8.73, the derivative of equation 8.72 with respect to the lift coefficient, corresponds to the lift coefficient that minimizes the fuel consumption.

$$\frac{d(\frac{1}{V}\frac{C_D}{C_L})}{dC_L} = \sqrt{\frac{\rho S}{2W}}\left(-\frac{1}{2}\frac{C_{D_o}}{\sqrt{C_L{}^3}} + \frac{3}{2}\frac{\sqrt{C_L}}{\pi Ae}\right) = 0 \tag{8.73}$$

The optimal cruise lift coefficient is then given by equation 8.74, i.e., the solution of equation 8.73.

$$C_L = \sqrt{\frac{C_{D_o}\pi Ae}{3}} \tag{8.74}$$

Using the parabolic drag polar approximation, the optimal cruise lift coefficient of equation 8.74 leads to equation 8.75, which gives an estimate of the aircraft drag coefficient in cruise, and to equation 8.76, which gives the drag to lift ratio for the greatest range. Equation 8.76 will be useful later when two different cruise schedules is studied in Sections 8.4 and 8.5. As far as the propagated error in the lift to drag ratio is concerned, the sources of inaccuracy come from the clean configuration parasite drag and the Oswald efficiency factor: their relative errors should be added up, but also divided by two due to the square root.

$$C_D = C_{D_o} + \frac{C_{D_o}}{3} = \frac{4}{3}C_{D_o} \tag{8.75}$$

$$\frac{C_D}{C_L} = \sqrt{\frac{16C_{D_o}}{3\pi Ae}} \tag{8.76}$$

All the variables appearing in equation 8.56 have been determined and, therefore, the weight fraction in cruise can be obtained by integration.

8.4 Weight Fraction for Cruise at Constant Speed

If the cruising speed and the lift coefficient are kept constant for the entire cruise, equation 8.71 shows that the aircraft weight to air density must also remain constant. In other words, the aircraft climbs as the fuel is burned off. Consequently, this cruise schedule is frequently called cruise-climb. Moreover, it also maximizes the aircraft range since the aircraft speed is kept at its maximum value as the weight decreases during the cruise. In fact, for other cruise schedules such as constant altitude and step-climb, the optimal speed decreases so that the rate of weight loss increases according to equation 8.56. For a given range, the largest cruise weight fraction is thus achieved for cruise-climb or, conversely, for a given cruise weight fraction, the corresponding range is maximum for cruise-climb at optimal cruise speed. Since the aircraft does not slow down in cruise-climb, it leads to a shorter flight duration, which is another advantage for the airline operations. When equation 8.56 is integrated along the range (horizontal) rather than the cruise-climb path (oblique), equation 8.77 approximates the cruise weight fraction.

$$\int_{W_{bc}}^{W_{ec}} \frac{dW}{W} = -\frac{1}{V_c} \int_0^R C_{TSFC} \frac{C_D}{C_L} dR \tag{8.77}$$

- W_{bc} : weight at the beginning of the cruise (lb)
- W_{ec} : weight at the end of the cruise (lb)
- R : range (nmi)

Assuming constant TSFC and lift coefficient for cruise, the right hand-side integrand is constant. After integration and manipulation, equation 8.78 is derived and gives the cruise-climb weight fraction.

$$\frac{W_{ec}}{W_{bc}} = e^{-\frac{C_{TSFC} R}{V_c} \frac{C_D}{C_L}} \tag{8.78}$$

[Raymer '92] and [Roskam & Lan] state that transport aircraft are normally not permitted to fly the cruise-climb in spite of the airlines'interests because air-traffic control restricts the aircraft to constant altitude cruises. However, this is the method that is suggested for analysing the cruise segment in [McCormick '79], [McCormick '95], [Raymer '92], [Roskam & Lan] and [Torenbeek].

8.5 Weight Fraction for Cruise at Constant Altitude

For a constant cruising altitude and lift coefficient, equation 8.71 shows that the cruising speed decreases as the square root of the aircraft weight. For this latter reason, the expression for the speed from equation 8.71 has been substituted in equation 8.56. Integrating along the range at constant altitude, equation 8.79 is obtained.

$$\int_{W_{bc}}^{W_{ec}} \frac{dW}{\sqrt{W}} = -\sqrt{\frac{\rho S}{2}} \int_0^R C_{TSFC} \frac{C_D}{\sqrt{C_L}} dR \tag{8.79}$$

In order to obtain the necessary weight fraction in cruise, the calculated integral, given in equation 8.80, still needs to be divided by the square root of the weight at the beginning of the cruise segment. This weight balances the lift, which is fully determined in equation 8.81 for a given cruising altitude, lift coefficient and initial cruising speed.

$$2\sqrt{W_{bc}} - 2\sqrt{W_{ec}} = C_{TSFC}\sqrt{\frac{\rho S}{2}}\frac{C_D}{\sqrt{C_L}}R \tag{8.80}$$

where

$$W_{bc} = \frac{1}{2}\rho {V_{bc}}^2 S C_L \tag{8.81}$$

and V_{bc} is the speed at the beginning of the cruise (knots)

Finally, equation 8.80 is divided by the square root of equation 8.81. Replacing the weight with the lift expression from equation 8.81 results, after simplification, in equation 8.82, which expresses the cruise weight fraction for a cruise at constant altitude.

$$\frac{W_{ec}}{W_{bc}} = (1 - \frac{C_{TSFC}R}{2V_{bc}}\frac{C_D}{C_L})^2 \tag{8.82}$$

8.6 Weight Fraction for Cruise with Stairsteps

Two different cruise schedules have been discussed in Sections 8.4 and 8.5. One might wonder which one should be preferred. [Raymer '92] writes that pilots may be permitted several "stairsteps" to a more optimal altitude during a long cruise. Referring to [Torenbeek], ATA '67 (Airline Transport Association) suggests for a basic flight profile analysis that there are no more than two step-climbs during a long-range cruise. According to commercial pilots themselves, except for long-range flights, they fly the entire cruise at the same constant altitude they have been assigned by the air traffic control. If their airspace is not too crowded, they may be invited to suggest an altitude to the controller, but it is generally not the case, especially in Europe. Usually, for inter-continental flights, a single stairstep is allowed.

Thus, it seems that the cruise at constant altitude is closer to daily operation and the result leads to a more conservative fuel estimate. However, the weight fractions from equations 8.78 and 8.82 have been compared and the following results were obtained. The maximum percent difference were 7% for a range of 6000 nautical miles and, as it could have been expected, this difference increased with increasing ranges. An intermediate and fast solution for taking into account a cruise with possible stairsteps would be to consider an average cruise weight fraction between the two estimates. An even better solution would be to divide the range into segments corresponding to the stairsteps. The weight fractions can then be calculated with equation 8.83 segment by segment for the stairstep altitudes.

$$\frac{W_{i+1}}{W_i} = (1 - \frac{C_{TSFC_i} \Delta R_i}{2V_i} \frac{C_D}{C_L})^2 \tag{8.83}$$

where i : step number

Since the weight to air density ratio is constant for the optimum cruise-climb, the weight fraction for a constant altitude segment determines the best altitude for the next segment. The pilot should suggest this altitude to the air traffic controller. Meanwhile, the new cruise speed can also be calculated from the next step altitude. The air density ratio variation with altitude in the troposhere and stratosphere are stated in equations 8.84 and 8.85 respectively.

$$\frac{\rho}{\rho_{sl}} = (1 - 6.875(10)^{-6}h)^{4.2561} \tag{8.84}$$

$$\frac{\rho}{\rho_{sl}} = .2971e^{-\frac{h-36089}{20806.7}} \tag{8.85}$$

Since the ratio of the density ratios after the stairstep and before the stairstep is equal to the corresponding weight fraction ($\frac{\rho_{i+1}}{\rho_i} = \frac{W_{i+1}}{W_i}$), equations 8.84 and 8.85 express the next step altitude as a function of the previous altitude and weight fraction. Equations 8.86 and 8.87 give the altitude for a step within the troposphere and the stratosphere, respectively. For a transition between troposphere and stratosphere, the stairstep has to be divided into two consecutive stairsteps: a first stairstep from the initial altitude to 36089 feet and a second one from 36089 feet to the final altitude. No range should be credited for the second step so that the second weight fraction should be equal to one.

$$h_{i+1} = \frac{1 - (1 - 6.875(10)^{-6} h_i)(\frac{W_{i+1}}{W_i})^{1/4.2561}}{6.875(10)^{-}6} \tag{8.86}$$

$$h_{i+1} = h_i + 20806.7 ln(\frac{W_{i+1}}{W_i}) \tag{8.87}$$

In this chapter, the cruise weight fraction was found. Based on current turbofan engine data, actual cruising altitudes and Mach numbers, the TSFC was assumed to be constant for the entire cruise. The cruising speed was obtained from the actual cruising Mach number and altitude rather than from optimum conditions. Indeed, the optimal cruising speed did not seem to match the cruising speed given in the references. The aerodynamic coefficients were chosen to maximize the range. So, from these assumptions, the cruise weight fraction was rederived from the references for the cruise at constant speed and developed for the cruise at constant altitude. This new equation, which seems to be closer to actual cruise conditions, was finally refined to take into account possible stairstep climbs.

Chapter 9

Loiter Weight Fraction

For estimating the fuel weight fraction, the cruise and loiter weight fractions are more likely to be influenced by the transport aircraft design than the weight fractions for other flight segments. The cruise weight fraction was the object of the preceding chapter. Here the loiter weight fraction is analyzed.

In [Torenbeek], ATA '67 specifies that reserve fuel for domestic and international operations (below and over 3000 nautical miles) should permit a loiter at 1500 feet for 45 and 30 minutes, respectively.

Equations 8.53 and 8.55 gave the rate of weight decrease as a function of the weight for jet aircraft and the thrust required for level flight, respectively. If these two relations are combined, equation 9.88 is obtained, which defines the instantaneous weight variation.

$$dW = -C_{TSFC}\frac{C_D}{C_L}W\,dt \tag{9.88}$$

9.1 TSFC in Loiter

Since the loiter altitude is fixed (1500 ft), the TSFC only depends on the loiter velocity. The turbofan engine JT9D-7A from [McCormick '79] has a range of TSFC between .5 and .7 for maximum cruise thrust at sea level. The TSFC of the PW4056 turbofan in [McCormick '95] varies between .4 and .6 for maximum cruise thrust at seal level. In both cases, the TSFC increases with the Mach number within these intervals. On the other hand, [Raymer '92] suggests a TSFC of .7 and .4 for low and high-bypass turbofan engines, respectively. Due to TSFC variation, the designer is again required to select the engine type to be installed and the loiter Mach number before estimating the loiter TSFC.

9.2 Drag to Lift Ratio in Loiter

The loiter segment, as it was the case for the cruise segment, is assumed to be flown at constant lift coefficient in [McCormick '79], [McCormick '95], [Raymer '92] and [Roskam & Lan]. Thus, the same assumption is made in the present analysis. The drag to lift ratio is then minimized with respect to the lift coefficient in order to minimize the loiter fuel consumption.

Since the criterion for loiter is to maximize loiter time rather than the range, it can be expected that the Mach number will be lower for loiter than for cruise. The lower altitude that is imposed during loiter causes the Mach number to decrease. So, the parabolic drag polar approximation is more justified here. Equation 9.89 expresses the drag to lift ratio for a parabolic drag polar approximation.

$$\frac{C_D}{C_L} = \frac{C_{D_o} + \frac{{C_L}^2}{\pi Ae}}{C_L} = \frac{C_{D_o}\pi Ae + {C_L}^2}{C_L \pi Ae} \tag{9.89}$$

It is easily checked that the positive root of equation 9.90, obtained by taking the derivative of equation 9.89 with respect to the lift coefficient, is the lift coefficient that minimizes the drag to lift ratio and consequently the fuel consumption.

$$\frac{d(\frac{C_D}{C_L})}{dC_L} = \frac{2{C_L}^2 - (C_{D_o}\pi Ae + {C_L}^2)}{{C_L}^2\pi Ae} = 0 \tag{9.90}$$

The solution of equation 9.90 is then the optimal loiter lift coefficient given in equation 9.91. Then, the parabolic drag polar approximation establishes the drag coefficient in equation 9.92. Finally, the best drag to lift ratio for loiter is calculated in equation 9.93. As far as the propagated error for the loiter lift to drag ratio is concerned, the remarks which apply to the lift to drag ratio in cruise remain same.

$$C_L = \sqrt{C_{D_o}\pi Ae} \tag{9.91}$$

$$C_D = 2C_{D_o} \tag{9.92}$$

$$\frac{C_D}{C_L} = \sqrt{\frac{4C_{D_o}}{\pi Ae}} \tag{9.93}$$

9.3 Loiter Weight Fraction Equation

Once the TSFC has been estimated and the drag to lift ratio is calculated, equation 9.88 can be integrated for loiter.

$$\int_{W_{bl}}^{W_{el}} \frac{dW}{W} = -\int_0^E C_{TSFC}\frac{C_D}{C_L}dt \tag{9.94}$$

- W_{bl} : weight at the beginning of the loiter (lb)
- W_{el} : weight at the end of the loiter (lb)
- E : endurance (hrs)

After having calculated the integral equation 9.94, the result has been rewritten with respect to the loiter weight fraction in equation 9.95.

$$\frac{W_{el}}{W_{bl}} = e^{-C_{TSFC} E \frac{C_D}{C_L}} \tag{9.95}$$

The loiter weight fraction was entirely rederived from the references. The endurance is stated by the ATA '67 requirements. The TSFC is assumed constant and some typical values of TSFC for current turbofan engines were listed. The optimal aerodynamic coefficients for loitering were calculated as a function of the wing aspect ratio.

Chapter 10

Gross Weight

"Weight minimization of an airplane design is a subject of the utmost importance." [Torenbeek]

The gross weight of the aircraft may be decomposed into three components : operating empty weight, fuel weight required for the design mission plus reserves, and the crew and payload weight for which the aircraft is designed:

$$W_g = W_e + W_p + W_f \tag{10.96}$$

where

- W_e : operating empty weight (lb)
- W_f : fuel weight (lb)
- W_p : crew and payload weight (lb)

It is observed in [Jane's 94-95] that for a maximum payload and fuel weight, the gross weight is usually greater than the maximum takeoff weight. The maximum useful load then implies a compromise (trade-off) between fuel and payload, i.e., between range and number of passengers. In the present analysis, the fuel weight is computed for a given mission, i.e., fixed design payload and range.

The reader should indeed remember that the gross weight prediction affects the wing planform area and the static thrust through the wing-loading and the static thrust over gross weight. So, it is important to obtain good estimates of the weights since the overall design relies on the gross weight accuracy.

10.1 Operating Empty Weight

The operating empty weight is defined as the weight of the aircraft without fuel, crew members, passengers, cargo and any baggage. It is composed of the airframe structure, propulsion system, operational items, airframe services and equipment. Statistical analysis is a convenient tool for determining the operating empty weight. Several statistical weight equations are found in the literature. Some of them give fast estimates while relying on a few essential parameters ([Raymer '92], [Torenbeek]) and yet others require an iterative process ([Raymer '92]). Other more accurate methods consist of adding the weights of the major aircraft components, but they depend upon a larger number of depending parameters that are often unknown at the conceptual design level ([Raymer '92], [Roskam 5], [Torenbeek]). However, for this work, the operating empty weight needs to be obtained as a function of known variables describing the aircraft concept and the design requirements as described in Chapter 2.

The references often do not list or describe the aircraft used for evaluating their statistical correlation. This practice should be discouraged for two reasons. One, the aircraft concept may be important : subsonic or supersonic ; number of decks ; short, medium or long range ; etc. In order to improve the accuracy, the statistical analysis must be applied to a particular family of aircraft. Two, within the same aircraft category, aircraft technology may have a significant influence. For instance, composite materials are slowly replacing traditional aluminum parts for building lighter airplanes. Fly-by-wire systems are also replacing traditional heavy mechanical devices. The analysis should take these trends into account and should frequently be updated with currently produced aircraft. Whenever references have not been published or revised recently, their equations should be used carefully.

10.1.1 Aircraft Database

The variables in the statistical operating empty weight equation are selected among the design requirements (defined in Chapter 2) and the aircraft variables. The aircraft variables are the design data and variables from Chapter 2 and all the intermediate variables used in the synthesis algorithm. The reader may wonder why the choice of the variables is restricted to this set. In fact, the design requirements and the aircraft variables are known at this stage of the design process and have a direct influence on the operating empty weight, whereas the other aircraft characteristics are still undetermined, e.g., wing thickness ratio, wing taper ratio, number of wheels, etc. Some of these known parameters may have a greater influence on the operating empty weight than others, but they can not be identified a priori. Only a careful analysis will determine which combination of these variables leads to a better approximation.

For a small number of mainly outdated aircraft, a rather complete set of data is found in aircraft manuals. But, it is difficult to gather comparable information for a significant number of current aircraft. Some variables are very difficult to find in the references, e.g., the maximum lift coefficient and the braking deceleration are rarely given. Other variables can be obtained for a limited number of aircraft, e.g., the approach speed, the FAR part 25 landing field length, range-payload curves, cruise altitude, etc. In general, books only quote the aircraft data which are relevant to their discussion. Moreover, comparing different references and even different editions, discrepancies among aircraft data are observed for identical aircraft (data were yearly corrected, maybe !). This database should contain the data for the major variables for an as large as possible number of current aircraft.

As one of the best aircraft data references, [Jane's 94-95] was selected to be the only source of information used. The possible variables that could be found frequently were the operating empty weights for several versions, the cabin and fuselage lengths and widths, the wing aspect ratio, a cruise altitude, the cruise Mach number, ranges and a landing field length. From this information, the cabin and fuselage fineness and the cruise speed could be calculated directly. The maximum landing weight fraction, the wing-loading and the static thrust over gross weight were computed from the maximum takeoff and landing weights, the wing surface area, and the static thrust, which were also available. However, not all these data were not available for every modern aircraft. Moreover, the maximum lift coefficient was never given in [Jane's 94-95], but it was obtained through the stall speed definition in landing configuration, equation 6.32, whenever the approach speed, the maximum landing weight and the wing surface area were given. The cruise and loiter weight fractions could not be found.

10.1.2 Statistical Model

A weight component analysis is unworkable with [Jane's 94-95] since the operating empty weight is not subdivided into weight components. Moreover, any combination of variables that can be used with the proposed design algorithm not capable of accurately approximating the different weight components.

The major weight components are usually modeled as multi-exponential approximations; i.e., products of a constant and exponentials of aircraft parameters. An identical formulation is assumed to approximate the operating empty weight in equation 10.97. A list of aircraft parameters is selected from among the available variables.

The factor C_0 and the exponents C_i in equation 10.97 will be calculated later such that the sum of the square of the operating empty weight error is minimized for every aircraft within the aircraft databank, i.e., a least square criterion. The quality of the approximations for different sets of variables is then be compared.

$$W_e = C_0 \Pi_{i=1}^{n} X_i{}^{C_i} \tag{10.97}$$

where

- C_i : i-th coefficient to be determined
- X_i : i-th variable

Many different combinations of variables have been tested, but, unfortunately, for different aircraft data sets. A typical data set would include all the aircraft from [Jane's 94-95] for which all the necessary variables are either given or could be calculated easily. In [Jane's 94-95], some values are not available for every aircraft so that the selection of a particular variable may drastically restrict the population size. On the other hand, as the ratio of the numbers of aircraft to variables decreases, the operating empty weight equation becomes more precise, but its ability to describe a larger population of data is uncertain. Therefore, the ultimate goal is to select a small number of variables for which a large number of aircraft can be used to determine the operating empty weight approximation resulting in the best accuracy. Thus, it is easily be understood that the choice of a particular operating empty weight model is based on a subjective judgement.

As expected, it was discovered that the aircraft concept strongly influences the operating empty weight. Important conceptual design characteristics that may justify the use of different operating empty weight equations are the number of engines, the engine mounting on the wing or the fuselage, the tail configuration, etc. So, as a rule of thumb, operating empty weight equations should be developed from data sets whose aircraft have the same overall shape, and their use should be restricted to similar concepts. A better accuracy is achieved if, for instance, different equations are developed separately for two, three or four engined aircraft. Whereas many different versions of twin engined transport aircraft have been produced recently, only a few three and four engine transport aircraft have been designed during the same period. A twin engined transport aircraft is used next.

10.1.3 Operating Empty Weight of Twin Turbofan Aircraft

In order to calculate the operating empty weight of a twin turbofan transport aircraft, many attempts were made and compared. One of the most succesful one is described here.

Since the major contributor to the operating empty weight is the fuselage, its dimensions must be taken into account. In chapter 3, it was shown that the cabin sizing is more accurate than the fuselage sizing. The cabin diameter is directly proportional to the number of seats abreast in economy class, which is one of the design data. The cabin length is a function of the number of passengers and seats abreast in first and economy classes. The range and the maximum landing to gross weight ratio characterizes the flight type : domestic or international. Long-range aircraft are frequently larger than short-range aircraft and have in general a smaller maximum landing to gross weight fraction. The wing aspect ratio, the maximum lift coefficient and the wing-loading define the wing. The wing aspect ratio and the maximum lift coefficient have to be selected for evaluating the operating empty weight since they are design data. They respectively describe the wing shape and the complexity of the high-lift devices. Together with the wing-loading, they implicitly define the engine thrust required since the takeoff is a compromise between large static thrust and efficient wing.

This particular choice of variables limited the aircraft data bank from [Jane's 94-95] to the 32 aircraft presented in Table 15 : 8 Airbus and 24 Boeing aircraft. In this table, the operating empty weight is measured in pounds, the cabin length and diameter in meters, the range in nautical miles and the wing-loading in pounds per square feet. The list is composed of medium to large-capacity commercial airliners designed for medium and long-range. These twin turbofan transport aircraft have wing mounted engines and a conventional tail configuration. The result of the analysis should predict closely the aircraft belonging to the same category and having a similar design.

The least square method was applied to the aircraft data in Table 15 and gives the factor and exponents of the operating empty weight model, equation 10.98. For the data in Table 15, the percent difference between the actual operating empty weight and its predicted value is less than 2% while the average is .67% (see appendix D).

$$W_e = 6556.45 {l_c}^{.7538} {w_c}^{1.07} R^{.244} {x_{ml}}^{-.344} A^{.2977} {C_{L_{max}}}^{.2762} \left(\frac{W_g}{S}\right)^{-.8475} \tag{10.98}$$

Although the actual aircraft parameters should yield an accurate operating empty weight using equation 10.98, the errors between actual and approximated aircraft parameters propagate throughout the calculation to produce a most likely less accurate result. More precisely, the corresponding relative errors are the sums of the relative error for each variable multiplied by its corresponding exponent. Luckily, except for the cabin width, these exponents are less than unity and the errors are decreased before being added. For the cabin width, the error is only slightly magnified.

It would be interesting to test the equation 10.98 for an aircraft that does not belong to the database used to obtain the approximation. However, it should not be tested with the actual aircraft parameters, but with their approximation from the design equations. The actual aircraft characteristics are indeed unknown and even irrelevant with respect to the conceptual design problem. For this reason, the testing

Aircraft	W_e	l_c	w_c	R	x_{ml}	A	$C_{L_{max}}$	W_g/S
A300-600	198665	40.21	5.28	3600	.836	7.73	2.973	129.98
A300-600R	199163	40.21	5.28	3950	.821	7.73	2.972	134.31
A300-600	198563	40.21	5.28	3600	.836	7.73	2.973	129.98
A300-600R	199000	40.21	5.28	4000	.821	7.73	2.972	134.31
A310-200	176683	33.24	5.28	3600	.866	8.80	3.146	132.80
A310-200	176645	33.24	5.28	3600	.866	8.80	3.146	132.80
A310-300	177128	33.24	5.28	4300	.820	8.80	3.146	140.29
A310-300	178225	33.24	5.28	4300	.820	8.80	3.146	140.29
B757-200	126060	36.09	3.53	2820	.900	7.82	2.841	110.33
B757-200	125750	36.09	3.53	2980	.900	7.82	2.841	110.33
B757-200	126060	36.09	3.53	3820	.792	7.82	2.841	125.38
B757-200	125750	36.09	3.53	4000	.792	7.82	2.841	125.38
B767-200	178400	33.93	4.72	3160	.900	7.99	2.386	98.360
B767-200	177500	33.93	4.72	3220	.900	7.99	2.386	98.360
B767-200	178400	33.93	4.72	3795	.863	7.99	2.403	103.28
B767-200	177500	33.93	4.72	3850	.863	7.99	2.403	103.28
B767-200ER	184200	33.93	4.72	5365	.806	7.99	2.386	113.11
B767-200ER	184000	33.93	4.72	5410	.806	7.99	2.318	113.11
B767-200ER	184700	33.93	4.72	6770	.736	7.99	2.376	126.89
B767-200ER	184500	33.93	4.72	6805	.736	7.99	2.376	126.89
B767-300	192100	40.36	4.72	4000	.870	7.99	2.466	113.11
B767-300	192100	40.36	4.72	4020	.870	7.99	2.466	113.11
B767-300	191700	40.36	4.72	4230	.855	7.99	2.466	115.08
B767-300	191700	40.36	4.72	4260	.855	7.99	2.466	115.08
B767-300ER	196900	40.36	4.72	5740	.775	7.99	2.466	126.89
B767-300ER	196500	40.36	4.72	5760	.775	7.99	2.466	126.89
B767-300ER	198200	40.36	4.72	6060	.800	7.99	2.487	131.15
B777-200	298900	48.97	5.87	3970	.879	8.68	2.529	109.88
B777-200	298900	48.97	5.87	4240	.864	8.68	2.529	111.83
B777-200	299550	48.97	5.87	4820	.832	8.68	2.529	116.18
B777-200	304500	48.97	5.87	6030	.793	8.68	2.540	125.95
B777-200	304500	48.97	5.87	6300	.780	8.68	2.540	128.12

Table 15: Operating empty weight and variables [Jane's 94-95]

of equation 10.98 is postponed to Section 11.2 where the complete algorithm is applied to a transport aircraft not in the database.

10.2 Crew and Payload Weight

The maximum crew and payload involves : pilots, cabin staff, passengers, baggage and cargo. The design payload weight can be expressed as equation 10.99.

$$W_p = n_c W_{fc} + ((n_f/n_{fa}) + (n_e/n_{ea}))W_a + n_f W_{fp} + n_e W_{ep} + W_c \tag{10.99}$$

where

- n_c : number of crew members
- W_{fc} : weight per flight crew member (lb)
- n_{fa} : number of first class passengers per flight attendant
- n_{ea} : number of economy class passengers per flight attendant
- W_a : weight per flight attendant (lb)
- W_{fp} : weight per first class passenger (lb)
- W_{ep} : weight per economy class passenger (lb)
- W_c : cargo weight (lb)

Whereas the number of flight crew members is usually specified within the design requirements together with the numbers of passengers in first and second classes, the number of cabin staff depends on the number of passengers per class and the type of flight : domestic or international. According to [Raymer '92] and [Torenbeek], there is a cabin staff member for every 16 to 20 first class passengers and every 31 to 36 economy class passengers. For high density cabin arrangement, there must be at least one cabin attendant for every 50 passengers.

Some guidelines for the weight are found in the references. The FAR part 25 requirements state that a minimum weight of 170 pounds per passenger should be taken into account. In [Raymer '92], each passenger with his carry-on baggage weighs an average of 180 pounds, while 40 to 60 pounds are to be added as checked-in baggage. For [Torenbeek], a passenger weighs about 170 pounds and the baggage weight per passenger is 35 and 40 pounds for short and long-range flights, respectively. The same reference also indicates 165 pounds per passsenger plus 40 and 60 pounds of baggage for tourist and first class, respectively. Flight and cabin crews with baggage and flight equipment weigh 205 and 150 pounds, respectively. In [Jane's 94-95], the weight per passenger ranges between 200 and 220 pounds. Finally, in addition to the baggage, cargo weight may or may not be among the design specifications, but no information could be found in the references.

10.3 Fuel Weight

The fuel weight is calculated for the entire mission from the design range and, after missed approach, flight to an alternate airport plus holding. Equation 10.100 is equation 10.96, where the fuel weight fraction has been substituted for convenience.

$$W_g = W_e + W_p + x_{fu} W_g \tag{10.100}$$

where x_{fu} : fuel weight fraction

Based on equation 10.100, the fuel weight can be expressed as equation 10.101.

$$W_f = x_{fu} W_g = x_{fu} \frac{W_e + W_p}{1 - x_{fu}} \tag{10.101}$$

In equation 10.101, all the terms have previously been determined except the fuel weight fraction. Equation 10.102 gives the fuel weight fraction for an airline mission including required reserves. The fuel weight fraction is expressed as a product of weight fractions for the different legs of the mission. A single fuel fraction is assumed for the engine start, warm-up, taxi and takeoff. According to ATA '67, climb and descent are to be counted twice : first, for the cruise and, second, for the flight to an alternate airport. The cruise weight fraction is calculated in chapter 8 : it should take into account the design range plus a portion of the 200 nautical miles to an alternate airport. As it has been explained in chapter 8, the aircraft flies at a cruising altitude and speed, a distance equal to the difference between the extra 200 nautical miles and the range transversed during for the second climb and descent. A short-cut approximation for the diversion to the alternate airport consists of adding 200 nautical miles to the design range rather than counting additional weight fractions for climb, detour at cruise conditions, and descent. The loiter weight fraction is discussed in chapter 9 and the landing, taxi and shut-off were gathered in a single constant weight fraction. Finally, the fuel weight fraction includes the unusable fuel trapped in the fuel system.

$$x_{fu} = x_{uf}(1 - x_{to}x_{cl}^2 x_{cr\&aa}x_{de}^2 x_{lo}x_{la}) \tag{10.102}$$

where

- x_{uf} : unusable fuel weight fraction
- x_{to} : start, warm-up, taxi and takeoff weight fraction
- x_{cl} : climb weight fraction
- $x_{cr\&aa}$: cruise and alternate airport diversion weight fraction
- x_{de} : descent weight fraction
- x_{lo} : loiter weight fraction
- x_{la} : landing, taxi and shutt-off weight fraction

Some references suggest average values for the constant weight fractions, but the way they were obtained was not indicated. Although some references have not been edited recently, it is expected that those values are still applicable to current technology aircraft. In [Roskam 1], the weight fractions for transport aircraft are 99% for engine starts and warm-up, 99% for taxi, 99.5% for takeoff, 98% for climb, 99% for descent and landing, 99.2% for taxi and shutt-off, respectively. In [Raymer '92], the global weight fraction for engine start, warm-up, taxi and takeoff varies from 97% to 99%, the climb weight ratio is 98.5%, and the landing weight fraction ranges between 99.2% and 99.7%. [Raymer '92] also indicates that about 6% of the total fuel is trapped in the fuel system.

For current transport aircraft equipped with two turbofan engines, the operating empty weight was statistically calculated from the cabin length and width, the range, the maximum landing weight fraction, the wing aspect ratio, the maximum lift coefficient and the wing-loading. The cabin length and width were sized in chapter 3 and the wing-loading was analyzed in chapter 6. The crew and payload weight was obtained from the respective numbers of passengers in first and economy classes. Finally, the operating empty weight, the crew and payload weight, the cruise weight fraction from chapter 8, and the loiter weight fraction from chapter 9 yielded the fuel weight.

Chapter 11

Synthesis Method

Different aspects of the conceptual design of transport aircraft are described in Chapters 3 to 10. In the present chapter, the design equations are assembled into a sequential algorithm. Then, the direct method is tested on a 'new' transport aircraft and the results are compared with the actual design.

11.1 Algorithm

The following algorithm was implemented in a computer program included in Appendix E.

11.1.1 Cabin and Fuselage Analysis

Step 1 : Cabin Width is a function of the number of seats abreast in economy class. Equations 3.1 and 3.2 both give the cabin width. Equation 3.1 can be used whenever the seat and aisle widths are specified or their approximate values are statistically estimated. Equation 3.2 is a linear correlation for two aisle cabins.

Step 2 : Fuselage Width depends on the cabin width. Equation 3.3 gives the fuselage width from the average fuselage thickness based on the references.

Step 3 : Number of Seats Abreast in First Class is obtained from the cabin width and the number of seats abreast in economy class. Equations 3.4 and 3.5 both give the number of seats abreast in first class. Equation 3.4 applies to the case where the seat and aisle widths are specified, while equation 3.5 gives identical results to the actual values in [Jane's 94-95] with statistically estimated widths.

Step 4 : Cabin Length is a function of the respective numbers of seats abreast and passengers in first and economy classes. Equations 3.6 and 3.7 both give the cabin length. Equation 3.6 can be used whenever the seat pitches and the other cabin dimensions (exit doors, galleys, toilet, etc) are specified. Otherwise, the statistical equation 3.7 gives better results.

Step 5 : Fuselage Length is computed from the cabin length. Equation 3.8 gives the fuselage length from the fuselage nose-tail cone distance. This latter distance is either a fraction of the fuselage width or an average distance obtained from current transport aircraft.

11.1.2 Performance Analysis

Step 6 : Stall Speed in Landing Configuration is calculated from the FAR part 25 landing field length and the constant braking deceleration. Equation 4.24 gives the stall speed in landing configuration. Typical values of the approach angle, load factor and stall speed factor for flare are chosen.

Step 7 : Stall Speed in Takeoff Configuration is a function of the stall speed in landing configuration and the maximum landing to gross weight ratio. Equation 5.29 gives the stall speed in takeoff configuration. It assumes constant maximum takeoff weight fraction and maximum lift coefficient fraction for takeoff configuration.

Step 8 : Wing-Loading depends on the stall speed in landing configuration, the maximum lift coefficient and the maximum landing over gross weight. Equation 6.30 gives the wing-loading. Air density at the airport is required.

Step 9 : Static Thrust over Gross Weight is obtained from the stall speeds in takeoff and landing configurations, the maximum landing over gross weight, the maximum lift coefficient and the wing aspect ratio. Equation 7.49 calculates the aircraft speed from the stall speed in the pertinent configuration. Equation 7.51 approximates the lift coefficient from its maximum value. Equation 7.52 estimates the drag coefficient from the lift coefficient and the wing aspect ratio. Equation 7.48 finally gives the corresponding thrust-to-weight ratio from the weight fraction, the aircraft speed, and the lift and drag coefficients. The stall speed fraction, the number of operating engines and the climb gradient are stated in the FAR part 25 climb requirements. These requirements also allow for an estimation of the maximum lift coefficient fraction, the parasite drag coefficient, the Oswald efficiency fraction, and the thrust fraction. The engine thrust characteristics were obtained for current turbofan engines. Equations 7.49, 7.51, 7.52 and 7.48 are successively applied to the second segment of climb and missed approach. The required static thrust over gross weight is the maximum of these two thrust-to-weight ratios.

11.1.3 Weight Analysis

Step 10 : Cruise Weight Fraction is calculated from the cruising altitude and Mach number, the range, and the wing aspect ratio. Equations 8.69 and 8.70 give the cruising speeds in the troposphere and stratosphere, respectively, as a function of the cruising altitude and Mach number. Equation 8.76 approximates the optimal drag-to-lift ratio in cruise from the wing aspect ratio. The parasite drag coefficient and the Oswald efficiency factor are assumed to be chosen. Equation 8.78 for a cruise at constant speed and equation 8.82 for a cruise at constant altitude finally give the cruise weight fraction from the cruising speed, the drag-to-lift ratio, and the range.

In the two preceding equations, the TSFC in cruise was chosen from updated engine performance data.

Step 11 : Loiter Weight Fraction is a function of the wing aspect ratio. Equation 9.93 approximates the optimal drag-to-lift ratio as a function of the wing aspect ratio. The parasite drag coefficient and the Oswald efficiency factor are assumed to be chosen. Equation 9.95 gives the loiter weight fraction from the drag-to-lift ratio. The endurance is stated in the ATA '67 regulations, and the TSFC in loiter is chosen from updated engine performance data.

Step 12 : Operating Empty Weight is computed from the cabin width and length, the range, the maximum landing weight fraction, the wing aspect ratio, the maximum lift coefficient, and the wing loading. Equation 10.98 gives a statistical estimation of the operating empty weight for current twin turbofan transport aircraft.

Step 13 : Crew and Payload Weight is a function of the respective numbers of passengers in first and economy classes. Equation 10.99 gives the crew and payload weight. It requires the weights per flight crew (pilots and cabin attendants) and the weights per passenger (first and economy classes) including baggage. The number of cabin attendants per passengers is taken from the references.

Step 14 : Fuel Weight depends on the operating empty weight, the crew and payload weight, the cruise weight fraction, and the loiter weight fraction. Equation 10.102 estimates the fuel weight fraction from the cruise and loiter weight fractions. Equation 10.101 gives the fuel weight from the operating empty weight, the crew and payload weight, and the fuel weight fraction.

Finally, the gross weight, the wing reference area, and the wing span are easily calculated.

11.2 Example

The algorithm is tested with a twin turbofan transport aircraft whose design requirements, data and results are found in the references. The method is validated, provided that the test aircraft has not been used for approximating the operating empty weight. As the operating empty weight analysis involved all the major western airliners whose necessary information were available, the choice of an example transport aircraft was limited to eastern conventional designs. If the method proves to be successful for different design practices and aviation regulations, the practical importance of this method will be emphasized.

11.2.1 Aircraft : Tu-204

All the necessary data could not be found for other aircraft in [Jane's 94-95]. However, the Tupolev 204 was selected because its description involved only two versions: the 200 and 300 series. It is a state-of-the-art twin-turbofan medium-range airliner. The 200 version deliveries started in 1995 and the 300 version was announced in 1994. Table 16 shows their respective characteristics. Whenever the 200 series data are available, they are used. In fact, the approach speed is not specified for the 300 series. It is the only characteristic assumed identical to the 300 series. The 200 series differ only by their engines. The Aviadvigatel PS-90P, Pratt & Whitney PW2240, and Rolls-Royce RB211-535 have a static thrust of 35580, 41700 and 43100 pounds, respectively. All the 300 series are equipped with Aviadvigatel engines (35580 lbs).

From Table 16, the reader should note that the aircraft was designed to fly a large range than is possible with its maximum payload. Paradoxically, the maximum takeoff weight is nearly equal to the sum of the operating empty weight, the maximum fuel weight and the maximum payload weight with space limited to 196 seats (see the value denoted with a star in Table 16). Since the design payload is not available, the aircraft will be designed to carry its maximum payload (see the value denoted

Tu-204	200 series	300 series
cabin length (ft)	99	N/A
cabin width (ft)	11.7	11.7
fuselage length (ft)	150.9	131.9
fuselage width (ft)	12.5	12.5
wing aspect ratio	9.67	N/A
wing area (ft^2)	1963.4	N/A
wing span (ft)	137.8	134.1
max T-O weight (lb)	244155	227070
operational empty weight (lb)	130070	N/A
max payload (lb)	43132*	39682
max fuel (lb)	72090	N/A
max landing weight (lb)	197310	N/A
nominal cruising altitude (ft)	36400-39700	36400-39700
nominal cruising speed (knots)	448	448-458
approach speed (knots)	N/A	122
range - max payload (nm)	3415	3585-3885
range - design payload (nm)	N/A	4075-4990

Table 16: Tu-204-200 and 300 [Jane's 94-95]

n_f	n_e	h_c (ft)	M_c	R (nm)	V_a (knots)
12	196	38050	.78	3415	122

Table 17: Test requirements [Jane's 94-95]

nsa_e	A	$C_{L_{max}}$	x_{ml}
6	9.67	3.366	.808

Table 18: Test design data [Jane's 94-95]

with a star in Table 16) over the corresponding range. For this version, the 196 seats are shared between 12 first class seats four-abreast at a pitch of 39 inches, and 184 economy class seats, six-abreast at a pitch of 31 inches.

11.2.2 Problem Statement

The test requirements were gathered in Table 17. The cruise Mach number was evaluated from the nominal cruising speed and the average nominal cruising altitude.

Assuming that the designer guessed the actual aircraft data for the design, Table 18 gives the corresponding design data. The maximum lift coefficient was calculated from the stall speed definition at maximum landing weight and sea level, equation 6.32, which is usually used for calculating the wing-loading. The maximum lift coefficient was obtained from the maximum landing weight (200 series), the wing area (200 series) and the approach speed (300 series). The maximum lift coefficient is relatively high compared to western transport aircraft, but the reader should be aware that the wing has a supercritical cross-section and is equipped with four-sections of double-slotted flaps and four-sections of leading edge slats over the full span of the wing. The actual maximum landing and maximum takeoff weights directly lead to the maximum landing weight fraction.

Test	calculated	actual
nsa_f	5	4
l_c (ft)	110.9	99.0
w_c (ft)	11.7	11.7
l_f (ft)	153.3	150.9
w_f (ft)	12.3	12.5
W_e (lb)	131381	130070
W_p (lb)	41580	43132
W_f (lb)	72368.6	72090
W_g (lb)	245329	244155
S (ft^2)	1972.21	1963.4
$T_{s/e}$ (lb)	37722	35580-43100

Table 19: Test design versus actual results [Jane's 94-95]

11.2.3 Design Results

From data in Table 17 and 18, the results in Table 19 were obtained using the computer program included in Appendix E.

The algorithm insures that first class passengers can be seated five abreast without discomfort. However, 12 first class passengers would require three rows (5-5-2) anyway. Instead, a four abreast seating (4-4-4) is a more convenient seating arrangement. The difference between calculated and actual cabin lengths seems to correspond to the galleys, buffets, and toilets. Indeed, the actual cabin length can barely contain the seating arrangement depicted in [Jane's 94-95], i.e., based on the given seat pitches and respective numbers of passengers in first and economy classes. On the other hand, the statistical correlation for calculating the cabin length takes into account the entire cabin length without the flight deck. For the static thrust, the calculated value is within the actual range. A possible reason for such a large range is that the Tu-204 performance was limited with the Russian Aviadvigatel engines. For a quick alternative, Tupolev would have had no choice other than buying existing power plants abroad, but available western engines would have been oversized for this aircraft.

First, the conceptual design algorithm was decomposed into successive steps. The equations used were described along with their required data and results. Then, the direct method was applied to the new Tupolev Tu-204. In spite of the fact that the design equations were derived from Western regulations and even though the statistical equations were obtained from Western aircraft data, the results for this eastern airliner were in complete agreement with the actual design.

Chapter 12

Conclusion

A conceptual design problem for transport aircraft was formulated in Chapter 2. The synthesis equations required for solving the problem were derived in Chapters 3 to 10. In Chapter 11, the previous equations were organized into a sequential algorithm and was succesfully tested on a new twin turbofan airliner.

The present conceptual problem is similar to general formulations with some differences. The distinction was made between passengers in first class and economy class, the cruising altitude was specified and the constant braking deceleration or the approach speed needed to be imposed. The designer's experience was required for selecting the number of seats abreast in economy class, the wing aspect ratio, the maximum lift coefficient and the maximum landing weight over gross weight. The FAR part 25 takeoff field length is to be checked a posteriori. The formulation differences enable to solve the problem by a direct sequential process.

In Chapter 3, single deck cabin and fuselage dimensions were sized with consideration towards current trends of seating arrangements. New statistical equations were obtained for estimating cabin and fuselage dimensions as a function of the respective numbers of passengers and seats abreast in first and economy classes. It was assumed that the cabin width depends only on the number of seats abreast in economy class.

Then, the number of seats abreast in the first class was calculated from the cabin width. The cabin length was calculated from the numbers of rows in the first and economy class compartments and the number of passengers. The fuselage width was obtained by adding an average fuselage thickness to the cabin width. Similarly, an average distance for the nose and tail cones had to be added to the cabin length in order to determine the fuselage length. For the current transport aircraft under consideration, the predictions using the new correlations were in better agreement than the previous equations published in the literature, as described in Chapter 3.

The performance was analyzed in Chapters 4 to 7. An expression of the stall speed in landing configuration was developed from a landing analysis in Chapter 4. It was expressed as a function of the landing field length and the braking deceleration. This latter parameter proved to be influenced by the landing conditions. In the following chapter, a stall speed in takeoff configuration was found and expressed from its value in landing configuration and the maximum landing weight over gross weight. The wing-loading was rederived in Chapter 6 from the stall speed definition. It seemed that more accurate wing-loading can be obtained if the stall speed definition is specified at the maximum landing weight. It is then a function of the maximum landing weight over the gross weight, the maximum lift coefficient, and the stall speed in landing configuration. The wing-loading equation was rewritten as a function of the maximum lift coefficient. Based on the actual wing-loading of existing aircraft, the calculated maximum lift coefficient corresponded to the actual type of high-lift devices installed on the aircraft. This testing permitted a validation of the wing-loading equation.

Considering the FAR part 25 climb requirements as the sizing criteria, the static thrust over gross weight was determined in Chapter 7. The equations of motions during climb were rederived while taking into account commonly neglected effects : thrust variation with speed, climb acceleration, weight fractions, etc. The corresponding climb equation depends on the stall speeds in takeoff and landing configurations, the maximum lift coefficient, the maximum landing weight over gross weight and the wing aspect ratio. The new method was later tested on a significant number of current transport aircraft whose actual static thrust over gross weight was available. It resulted in a more accurate estimate of the static thrust over gross weight than could be obtained with the simplified models from the references considered.

The weight fractions in cruise and loiter were treated in Chapters 8 and 9. An expression of the best cruising altitude was derived as a function of the design cruising Mach number. But, the results did not match actual cruising speed and altitude for the transport aircraft. So, a specified cruising altitude was assumed since, in practice, it is imposed by the air traffic control. Rather than using the cruise-climb weight fraction suggested in the literature, a weight fraction for cruise at constant altitude was derived. The discrepancies between the constant speed and altitude models proved to be significant, especially for long-range airliners. The new expression for the cruise weight fraction was then modified in order to account for possible stairsteps. The loiter weight fraction expression, which requires only the wing aspect ratio, was rederived from the references.

In Chapter 10, the gross weight was resolved into the operating empty weight, the crew and payload weight, and the fuel weight. A statistical method was presented for estimating the operating empty weight. The statistical analysis was applied to current twin turbofan transport aircraft for which the necessary data were found. A particular expression that was chosen for the operating empty weight is a function of the cabin width and length, the range, the maximum landing weight over gross weight, the wing aspect ratio, the maximum lift coefficient, and the wing-loading. It led to a very accurate prediction for medium and large capacity and range airliners with conventional tails and wing mounted engines. The payload and crew weight was estimated from the numbers of passengers in first and economy classes, while the weights per crew member and passenger were found in the references. The fuel weight is calculated from the two previous weight components and the weight fractions relevant to the flight segment.

Finally, in Chapter 11, the preceding design equations were organized in a sequential algorithm for synthesizing the transport aircraft at the conceptual design level. This new direct method was tested succesfully on the design of the Tu-204, a potentially difficult problem to solve since the design criteria and the correlations used were based exclusively on western aircraft.

Suggestions involve the testing of the algorithm with other existing transport aircraft since only an extensive testing over other well-defined aircraft will permit a better evaluation of the algorithm's strengths and weaknesses. However, other trial problems could not be found since the aircraft data were incomplete in the references, and the aircraft used for the statistical analyses had to be disregarded. The present study and the final algorithm were conceived from the information that were accessible. If more aircraft are available, the statistical equations should also be refined and the extension of the method to three and four engines aircraft and multiple deck cabin designs would be beneficial. Since the method is a closed form sequential algorithm, improvements of the algorithm seem difficult. The optimization of the conceptual design was successfully implemented using the design algorithm described in this thesis (see computer programs in Appendix F). It should be noted, however, that different statistical models for estimating the operating empty weight were tested and gave different optimum concepts. This indicates that the optimization is sensitive to the statistical equation which is employed. So, the results should be interpreted with great care and further research should focus on improving the operating empty weight equation.

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Appendix A

Cabin Length Approximation

A.1 Computer Program : MLA.f

```
PROGRAM MLA

C written by FRUYTIER Pierre-Andre (1995)

IMPLICIT NONE

INTEGER M, ! aircraft number
& N ! variables number

PARAMETER (M=22,N=4)

INTEGER I,J,K ! indices

INTEGER X(1:M,0:N)

REAL SUM, ! sum
```

```
& F, ! estimated cabin length
& E, ! relative error
& ME ! mean error

REAL A(0:N,0:N),
& B(0:N),
& Y(1:M)


C 1) Reading of the data from a file

OPEN(UNIT=1,FILE='CABIN.DAT',FORM='FORMATTED')

DO I=1,M
  READ (1,*) X(I,0),Y(I),(X(I,J),J=1,N)
END DO

DO I=1,M
  X(I,1)=(X(I,3)/X(I,1))
  X(I,2)=(X(I,4)/X(I,2))
  X(I,3)=X(I,3)+X(I,4)
END DO

CLOSE(UNIT=1)

C 2) Computation of the A(I,J) and B(I) coefficients

DO I=0,N-1
  B(I)=0.
```

```
  DO J=I,N-1
    A(I,J)=0.
  END DO
END DO

A(0,0)=M
DO K=1,M
  B(0)=B(0)+Y(K)
  DO I=1,N-1
    B(I)=B(I)+Y(K)*X(K,I)
    A(0,I)=A(0,I)+X(K,I)
    DO J=I,N-1
      A(I,J)=A(I,J)+X(K,I)*X(K,J)
    END DO
  END DO
END DO

DO I=0,N-1
  DO J=I,N-1
    A(J,I)=A(I,J)
  END DO
END DO

C 3) Solution of the linear system of equations

DO J=0,N-1
  DO I=1,N-1
    SUM=A(I,J)
    IF (I.LE.J) THEN
```

```
      DO K=0,I-1
SUM=SUM-A(I,K)*A(K,J)
      END DO
      A(I,J)=SUM
    ELSE
IF (J.NE.0) THEN
  DO K=0,J-1
    SUM=SUM-A(I,K)*A(K,J)
  END DO
END IF
A(I,J)=SUM/A(J,J)
    END IF
  END DO
END DO

DO I=1,N-1
  SUM=B(I)
  DO J=0,I-1
    SUM=SUM-A(I,J)*B(J)
  END DO
  B(I)=SUM
END DO
DO I=N-1,0,-1
  SUM=B(I)
  DO J=N-1,I,-1
    IF (I.NE.J) THEN
    SUM=SUM-A(I,J)*B(J)
    END IF
  END DO
```

```
  B(I)=SUM/A(I,I)
END DO

PRINT*,'-* Solutions *-'
PRINT*,''
PRINT*,'C(0)=',B(0)
DO I=1,N-1
  PRINT*,'C(',I,')=',B(I)
END DO
PRINT*,''

C 4) Evaluation of the percent difference for each aircraft

PRINT*,'Aircraft type | % difference'
ME=0.
DO K=1,M
  F=B(0)
  DO I=1,N-1
    F=F+X(K,I)*B(I)
  END DO
  E=(Y(K)-F)/Y(K)*100
  ME=ME+ABS(E)
  PRINT*, X(K,0),'     |       ',E
END DO
ME=ME/M
PRINT *,''
PRINT *,' Mean error (%) =',ME
PRINT *,''
```

```
END
```

A.2 File : CABIN.DAT

```
300.601 131.9 6 8 26 240
300.602 131.9 1 8 0 289
310.1 109 6 8 20 200
310.2 109 1 9 0 280
320.1 89.8 4 6 12 138
320.2 89.80 1 6 0 164
321 112.8 4 6 16 169
737.301 77.2 4 6 8 120
737.302 77.2 1 6 0 141
737.303 77.2 1 6 0 149
757.1 118.4 4 6 16 162
757.2 118.4 4 6 16 170
757.3 118.4 4 6 12 190
757.4 118.4 4 6 12 196
757.5 118.4 1 6 0 214
757.6 118.4 1 6 0 220
757.7 118.4 1 6 0 223
757.8 118.4 1 6 0 224
767.201 111.3 6 7 18 198
767.202 111.3 1 7 0 230
767.203 111.3 1 7 0 242
767.204 111.3 1 8 0 255

#aircraft LC NSA1 NSA2 NP1 NP2
```

```
-* Solutions *-

C(0)=    0.872511
C(  1)=     5.49028
C(  2)=     2.41578
C(  3)=    0.134680

Aircraft type    |       % difference
  300      |        0.582240
  300      |         3.89443
  310      |         1.49767
  310      |        -4.10294
  320      |        -3.68414
  320      |         1.79712
  321      |        -2.29730
  737      |       -0.269144
  737      |         2.29868
  757      |         5.37770
  757      |         2.42733
  757      |       -0.876678
  757      |        -3.59953
  757      |         3.50807
  757      |        0.785203
  757      |        -1.59641
  757      |        -1.71016
  767      |        -2.49430
  767      |         1.92813
  767      |        -3.86497
  767      |         1.07350
```

```
 737      |          -2.22622

Mean error (%) =    2.35872
```

Appendix B

Turbofan Characteristic Approximation

B.1 Computer Program : avgthrust.f

```
PROGRAM AVGTHRUST

C written by FRUYTIER Pierre-Andre (1995)

IMPLICIT NONE

REAL M, ! Mach number
& V, ! speed
& T1,T2,T3,T ! static thrust

OPEN(UNIT=0,FILE='THRUST.DAT',STATUS='UNKNOWN')

PRINT *, 'V(ft/s)     | DT1(%)     | DT2(%)     | DT3(%)'
M=0.
```

```
DO WHILE (M.LE.0.26)
  V=1116.4*M
  T1=1-1.013E-3*V+1.013E-6*V**2 ! JT9D-7A
  T2=1-1.008E-3*V+.666E-6*V**2  ! Raymer
  T3=1-.827E-3*V+.642E-6*V**2 ! PW4056
  T=(T1+T2+T3)/3
  WRITE (0,20) T, V, V**2
  PRINT 10, V, (1-T1/T)*100, (1-T2/T)*100, (1-T3/T)*100
  M=M+.01
END DO

10 FORMAT(E11.4,1X,E11.4,1X,E11.4,1X,E11.4)
20 FORMAT(E11.4,1X,E11.4,1X,E11.4)

END
```

B.2 Computer Program : quadthrust.f

```
PROGRAM QUADTHRUST

C written by FRUYTIER Pierre-Andre (1995)

IMPLICIT NONE

INTEGER M, ! speed increment
& N ! variables number

PARAMETER (M=25,N=2)
```

```
INTEGER I,J,K ! indices

REAL X(1:M,0:N)

REAL SUM, ! sum
& F, ! estimated non-dimensional thrust
& E, ! relative error
& ME ! mean error

REAL A(0:N,0:N),
& B(0:N),
& Y(1:M)


C 1) Reading of the data from a file

OPEN(UNIT=1,FILE='THRUST.DAT',FORM='FORMATTED')

DO I=1,M
  READ (1,*) Y(I),(X(I,J),J=1,N)
END DO

CLOSE(UNIT=1)

C 2) Computation of the A(I,J) and B(I) coefficients

DO I=0,N
  B(I)=0.
  DO J=I,N
```

```
    A(I,J)=0.
  END DO
END DO

A(0,0)=M
DO K=1,M
  B(0)=B(0)+Y(K)
  DO I=1,N
    B(I)=B(I)+Y(K)*X(K,I)
    A(0,I)=A(0,I)+X(K,I)
    DO J=I,N
      A(I,J)=A(I,J)+X(K,I)*X(K,J)
    END DO
  END DO
END DO

DO I=0,N
  DO J=I,N
    A(J,I)=A(I,J)
  END DO
END DO

C 3) Solution of the linear system of equations

DO J=0,N
  DO I=1,N
    SUM=A(I,J)
    IF (I.LE.J) THEN
      DO K=0,I-1
```

```
SUM=SUM-A(I,K)*A(K,J)
      END DO
      A(I,J)=SUM
    ELSE
IF (J.NE.0) THEN
  DO K=0,J-1
    SUM=SUM-A(I,K)*A(K,J)
  END DO
END IF
A(I,J)=SUM/A(J,J)
    END IF
  END DO
END DO

DO I=1,N
  SUM=B(I)
  DO J=0,I-1
    SUM=SUM-A(I,J)*B(J)
  END DO
  B(I)=SUM
END DO
DO I=N,0,-1
  SUM=B(I)
  DO J=N,I,-1
    IF (I.NE.J) THEN
    SUM=SUM-A(I,J)*B(J)
    END IF
  END DO
  B(I)=SUM/A(I,I)
```

```
END DO

PRINT*,'-* Solutions *-'
PRINT*,''
PRINT*,'C(0)=',B(0)
DO I=1,N
  PRINT*,'C(',I,')=',B(I)
END DO
PRINT*,''

C 4) Evaluation of the percent difference for each speed

PRINT*,'speed (ft/s) | % difference'
ME=0.
DO K=1,M
  F=B(0)
  DO I=1,N
    F=F+X(K,I)*B(I)
  END DO
  E=(Y(K)-F)/Y(K)*100
  ME=ME+ABS(E)
  PRINT*, X(K,1),'     |      ',E
END DO
ME=ME/M
PRINT *,''
PRINT *,' Mean error (%) =',ME
PRINT *,''

END
```

B.3 File : THRUST.DAT

0.1000E+01	0.0000E+00	0.0000E+00
0.9895E+00	0.1116E+02	0.1246E+03
0.9792E+00	0.2233E+02	0.4985E+03
0.9691E+00	0.3349E+02	0.1122E+04
0.9591E+00	0.4466E+02	0.1994E+04
0.9494E+00	0.5582E+02	0.3116E+04
0.9399E+00	0.6698E+02	0.4487E+04
0.9305E+00	0.7815E+02	0.6107E+04
0.9214E+00	0.8931E+02	0.7977E+04
0.9124E+00	0.1005E+03	0.1010E+05
0.9037E+00	0.1116E+03	0.1246E+05
0.8951E+00	0.1228E+03	0.1508E+05
0.8867E+00	0.1340E+03	0.1795E+05
0.8785E+00	0.1451E+03	0.2106E+05
0.8705E+00	0.1563E+03	0.2443E+05
0.8627E+00	0.1675E+03	0.2804E+05
0.8551E+00	0.1786E+03	0.3191E+05
0.8477E+00	0.1898E+03	0.3602E+05
0.8405E+00	0.2010E+03	0.4038E+05
0.8334E+00	0.2121E+03	0.4499E+05
0.8266E+00	0.2233E+03	0.4985E+05
0.8200E+00	0.2344E+03	0.5496E+05
0.8135E+00	0.2456E+03	0.6032E+05
0.8072E+00	0.2568E+03	0.6593E+05
0.8012E+00	0.2679E+03	0.7179E+05
0.7953E+00	0.2791E+03	0.7790E+05
T(V)/Ts(-)	V(ft/s)	V^2(ft/s)^2

```
-* Solutions *-

C(0)=    0.999994
C(  1)=    -9.49242E-04
C(  2)=     7.73296E-07
```

speed(ft/s)	% difference
0.	6.02007E-04
11.1600	3.19257E-04
22.3300	1.75308E-03
33.4900	2.93995E-03
44.6600	-4.46211E-03
55.8200	-1.77671E-03
66.9800	1.75028E-03
78.1500	-3.57436E-03
89.3100	1.54607E-03
100.5000	-5.94479E-04
111.600	6.79349E-04
122.800	1.30516E-03
134.000	2.68211E-03
145.100	-5.06826E-03
156.300	-2.19110E-03
167.500	2.41818E-03
178.600	-4.11956E-03
189.800	2.13753E-03
201.000	9.27577E-03
212.100	-6.03628E-03
223.300	2.78338E-03

234.400	|	9.81296E-04
245.600	|	-6.59425E-04
256.800	|	-1.48421E-03
267.900	|	-8.70412E-04

Mean error (%) = 2.48041E-03

Appendix C

Static Thrust over Gross Weight Analysis

C.1 Computer Program : TW.f

```
PROGRAM TW

C written by FRUYTIER Pierre-Andre (1995)

IMPLICIT NONE

INTEGER N ! aircraft number

REAL PI, ! pi constant (-)
& E, ! Oswald number (-)
& TV, ! V coefficient for thrust @ sea level (lb*s/ft)
& TV2, ! V2 coefficient for thrust @ sea level (lb*s2/ft2)
& WTOG, ! weight fraction after take-off (-)
& C, ! acceleration @ climb (s2/ft2)
```

```
& CDOC, ! CD0 clean (-)
& FSCG, ! 1st segment of climb gradient (-)
& SSCG, ! 2nd segment of climb gradient (-)
& TSCG, ! 3rd segment of climb gradient (-)
& MAG, ! missed approach gradient (-)
& MLG ! missed landing gradient (-)


PARAMETER (N=23,
& PI=3.141592,
& E=.7,
& TV=-.949E-3,
& TV2=.773E-6,
& WTOG=.97,
& C=0.,
& CDOC=.016,
& FSCG=0.,
& SSCG=.024,
& TSCG=.012,
& MAG=.021,
& MLG=.032)

INTEGER I ! iterator

C aircraft parameters

REAL AR, ! aspect ratio (-)
& CLM, ! CL-maximum (-)
& VA, ! approach speed (knots)
```

```
& MWLG ! maximum landing weight fraction (-)

C models variables

REAL VSL, ! stall speed @ landing (ft/s)
& VSTO, ! stall speed @ take-off (ft/s)
& V, ! velocity (ft/s)
& VA, ! approach speed (ft/s)
& CDCL ! CD/CL (-)

C T/W

REAL TWFSC, ! T/W @ first segment of climb (-)
& TWSSC, ! T/W @ second segment of climb (-)
& TWSSCL, ! T/W @ first segment of climb (Loftin) (-)
& TWTSC, ! T/W @ third segment of climb (-)
& TWMA, ! T/W @ missed approach (-)
& TWMAL, ! T/W @ missed approach (Loftin) (-)
& TWML, ! T/W @ missed landing (-)
& TW ! T/W - maximum (-)

CHARACTER*16 NAME

OPEN(UNIT=0,FILE='TW.OUT',STATUS='UNKNOWN')
OPEN(UNIT=1,FILE='TW.IN',FORM='FORMATTED')

WRITE (0,5)

DO I=1,N
```

```
READ (1,*) NAME,AR,CLM,VA,MWLG

C Approach vs Landing -> VSL

VSL=VA*1.689/1.3

C PRINT*,'Vs@l (ft/s) =',VSL

C Landing vs Take-off -> VSTO

VSTO=VSL*SQRT(WTOG/MWLG/.8)

C PRINT*,'Vs@to (ft/s) =',VSTO

C First-segment of climb -> TW

V=1.1*VSTO
CDCL=(CDOC+.04+(.8*CLM/1.1**2)**2/PI/AR/E)/(.8*CLM/1.1**2)

TWFSC=2*WTOG/(1+TV*V+TV2*V**2)*(CDCL+FSCG*(1+C*V**2))

C PRINT*,'T/W 1st segment of climb (-) =',TWFSC

C Second segment of climb -> TW

V=1.2*VSTO
CDCL=(CDOC+.018+(.8*CLM/1.2**2)**2/PI/AR/E)/(.8*CLM/1.2**2)
```

```
TWSSC=2*WTOG/(1+TV*V+TV2*V**2)*(CDCL+SSCG*(1+C*V**2))
TWSSCL=2*(CDCL+SSCG)

C PRINT*,'T/W 2nd segment of climb (-) =',TWSSC
C PRINT*,'T/W (Loftin''s value) (-) =',TWSSCL

C Third segment of climb -> TW

V=1.25*VSTO
CDCL=(CD0C+(.65*CLM/1.25**2)**2/PI/AR/E)/(.65*CLM/1.25**2)

TWTSC=2*WTOG/.867/(1+TV*V+TV2*V**2)*(CDCL+TSCG*(1+C*V**2))

C PRINT*,'T/W 3rd segment of climb (-) =',TWTSC

C Go-around in approach configuration (Missed Approach) -> TW

V=1.5*1.1*VSL
CDCL=(CD0C+.045+(CLM/(1.5*1.1)**2)**2/PI/AR/E)/(CLM/(1.5*1.1)**2)

TWMA=2*MWLG/(1+TV*V+TV2*V**2)*(CDCL+MAG*(1+C*V**2))
TWMAL=2*(CDCL+MAG)

C PRINT*,'T/W missed approach (-) =',TWMA
C PRINT*,'T/W (Loftin''s value) (-) =',TWMAL

C Go-around in landing configuration (Missed Landing) -> TW

V=1.3*VSL
```

```
CDCL=(CDOC+.09+(CLM/1.3**2)**2/PI/AR/E)/(CLM/1.3**2)

TWML=MWLG/(1+TV*V+TV2*V**2)*(CDCL+MLG*(1+C*V**2))

C PRINT*,'T/W missed landing (-) =',TWML

C TW estimate

TW=MAX(TWFSC,TWSSC,TWTSC,TWMA,TWML)

C PRINT*,'=> T/W (-) =',TW

WRITE (0,10) NAME,AR,MWLG,TWFSC,TWSSC,TWSSCL,TWTSC,TWMA,TWMAL,TWML,TW

END DO

5 FORMAT('NAME                AR   MWLG  TWFSC TWSSC TWSSCLTWTSC TWMA
&  TWMAL TWML  TW')
10 FORMAT(A,F7.2,F6.3,8F6.3)

END
```

C.2 File : TW.IN

```
A300-600 7.73 2.973 135 .836
A300-600R 7.73 2.972 136 .821
A310-200 8.80 3.146 135 .866
A310-200(opt) 8.80 3.172 135 .866
A310-300 8.80 3.146 135 .820
```

```
A310-300(opt) 8.80 3.172 135 .820
B-757-200 7.82 2.841 132 .860
B-757-200 7.82 2.841 132 .792
B-767-200(1) 7.99 2.386 136 .900
B-767-200(2) 7.99 2.403 136 .863
B-767-200(3) 7.99 2.386 138 .806
B-767-200(4) 7.99 2.318 140 .806
B-767-200(5) 7.99 2.318 140 .736
B-767-300(1) 7.99 2.466 141 .870
B-767-300(1) 7.99 2.466 141 .855
B-767-300(1) 7.99 2.466 141 .775
B-767-300(2) 7.99 2.487 145 .800
B-777-200(A1) 8.68 2.529 138 .879
B-777-200(A2) 8.68 2.529 138 .864
B-777-200(A3) 8.68 2.529 138 .832
B-777-200(B1) 8.68 2.540 140 .793
B-777-200(B2) 8.68 2.540 140 .780
Tu-204 9.67 3.366 122 .808
# aircraft AR(-) CLM(-) VA(kts) MWLG(-)
```

C.3 File : TW.OUT

```
NAME             TWFSC TWSSC TWSSCLTWTSC TWMA  TWMAL TWML  TW
A300-600         0.340 0.340 0.283 0.272 0.299 0.282 0.199 0.340
A300-600R        0.341 0.341 0.283 0.273 0.294 0.282 0.195 0.341
A310-200         0.317 0.320 0.268 0.255 0.293 0.267 0.195 0.320
A310-200(opt)    0.318 0.321 0.269 0.256 0.293 0.267 0.195 0.321
A310-300         0.318 0.321 0.268 0.256 0.277 0.267 0.184 0.321
A310-300(opt)    0.320 0.323 0.269 0.258 0.277 0.267 0.185 0.323
```

B-757-200	0.326	0.327	0.275	0.261	0.304	0.280	0.201	0.327
B-757-200	0.329	0.330	0.275	0.263	0.280	0.280	0.185	0.330
B-767-200(1)	0.295	0.298	0.250	0.234	0.320	0.281	0.205	0.320
B-767-200(2)	0.296	0.300	0.251	0.236	0.307	0.281	0.197	0.307
B-767-200(3)	0.298	0.302	0.250	0.237	0.288	0.281	0.184	0.302
B-767-200(4)	0.295	0.299	0.247	0.235	0.290	0.282	0.184	0.299
B-767-200(5)	0.297	0.302	0.247	0.237	0.265	0.282	0.168	0.302
B-767-300(1)	0.302	0.305	0.254	0.241	0.311	0.280	0.200	0.311
B-767-300(1)	0.302	0.306	0.254	0.241	0.305	0.280	0.196	0.306
B-767-300(1)	0.305	0.308	0.254	0.243	0.277	0.280	0.178	0.308
B-767-300(2)	0.306	0.310	0.254	0.245	0.287	0.280	0.185	0.310
B-777-200(A1)	0.286	0.292	0.244	0.229	0.302	0.271	0.194	0.302
B-777-200(A2)	0.286	0.292	0.244	0.230	0.297	0.271	0.191	0.297
B-777-200(A3)	0.287	0.293	0.244	0.230	0.286	0.271	0.184	0.293
B-777-200(B1)	0.290	0.296	0.244	0.232	0.273	0.271	0.176	0.296
Tu-204	0.302	0.308	0.260	0.245	0.258	0.257	0.173	0.308

Appendix D

Operating Empty Weight Analysis

D.1 Computer Program : MEA.f

```
PROGRAM MEA

C written by FRUYTIER Pierre-Andre (1994)

IMPLICIT NONE

INTEGER M, ! aircraft number
& N ! variables number

PARAMETER (M=32,N=7)

INTEGER I,J,K ! indices

REAL SUM, ! sum
& F, ! estimated weight
& E, ! relative error
```

```
& ME ! mean error

REAL A(0:N,0:N),
& B(0:N),
& X(1:M,0:N),
& Y(1:M)

C 1) Reading of the data from a file

OPEN(UNIT=1,FILE='DATA.IN',FORM='FORMATTED')

DO I=1,M
  READ (1,*) X(I,0),Y(I),(X(I,J),J=1,N)
END DO

CLOSE(UNIT=1)

C 2) Computation of the A(I,J) and B(I) coefficients

DO I=0,N
  B(I)=0.
  DO J=I,N
    A(I,J)=0.
  END DO
END DO

A(0,0)=M
DO K=1,M
  B(0)=B(0)+LOG(Y(K))
```

```
  DO I=1,N
    B(I)=B(I)+LOG(Y(K))*LOG(X(K,I))
    A(0,I)=A(0,I)+LOG(X(K,I))
    DO J=I,N
      A(I,J)=A(I,J)+LOG(X(K,I))*LOG(X(K,J))
    END DO
  END DO
END DO

DO I=0,N
  DO J=I,N
    A(J,I)=A(I,J)
  END DO
END DO

C 3) Solution of the linear system of equations

DO J=0,N
  DO I=1,N
    SUM=A(I,J)
    IF (I.LE.J) THEN
      DO K=0,I-1
SUM=SUM-A(I,K)*A(K,J)
      END DO
      A(I,J)=SUM
    ELSE
IF (J.NE.0) THEN
  DO K=0,J-1
    SUM=SUM-A(I,K)*A(K,J)
```

```
  END DO
END IF
A(I,J)=SUM/A(J,J)
    END IF
  END DO
END DO

DO I=1,N
  SUM=B(I)
  DO J=0,I-1
    SUM=SUM-A(I,J)*B(J)
  END DO
  B(I)=SUM
END DO
DO I=N,0,-1
  SUM=B(I)
  DO J=N,I,-1
    IF (I.NE.J) THEN
    SUM=SUM-A(I,J)*B(J)
    END IF
  END DO
  B(I)=SUM/A(I,I)
END DO

B(0)=EXP(B(0))

PRINT*,'-* Solutions *-'
PRINT*,''
PRINT*,'C(0)=',B(0)
```

```
DO I=1,N
  PRINT*,'C(',I,')=',B(I)
END DO
PRINT*,''

C 4) Evaluation of the percent difference for each aircraft

PRINT*,'Aircraft type | % difference'
ME=0.
DO K=1,M
  F=B(0)
  DO I=1,N
    F=F*X(K,I)**B(I)
  END DO
  E=(Y(K)-F)/Y(K)*100
  ME=ME+ABS(E)
  PRINT*, X(K,0),'     |       ',E
END DO
ME=ME/M
PRINT *,''
PRINT *,' Mean error (%) =',ME
PRINT *,''

END
```

D.2 File : DATA.IN

```
300.61 198665. 40.21 5.28 7.73 3600. .836 2.973 129.98
300.62 199163. 40.21 5.28 7.73 3950. .821 2.972 134.31
```

300.63	198563.	40.21	5.28	7.73	3600.	.836	2.973	129.98
300.64	199000.	40.21	5.28	7.73	4000.	.821	2.972	134.31
310.21	176683.	33.24	5.28	8.80	3600.	.866	3.146	132.80
310.22	176645.	33.24	5.28	8.80	3600.	.866	3.146	132.80
310.31	177128.	33.24	5.28	8.80	4300.	.820	3.146	140.29
310.32	178225.	33.24	5.28	8.80	4300.	.820	3.146	140.29
757.1	126060.	36.09	3.53	7.82	2820.	.900	2.841	110.33
757.2	125750.	36.09	3.53	7.82	2980.	.900	2.841	110.33
757.3	126060.	36.09	3.53	7.82	3820.	.792	2.841	125.38
757.4	125750.	36.09	3.53	7.82	4000.	.792	2.841	125.38
767.21	178400.	33.93	4.72	7.99	3160.	.900	2.386	98.360
767.22	177500.	33.93	4.72	7.99	3220.	.900	2.386	98.360
767.23	178400.	33.93	4.72	7.99	3795.	.863	2.403	103.28
767.24	177500.	33.93	4.72	7.99	3850.	.863	2.403	103.28
767.25	184200.	33.93	4.72	7.99	5365.	.806	2.386	113.11
767.26	184000.	33.93	4.72	7.99	5410.	.806	2.318	113.11
767.27	184700.	33.93	4.72	7.99	6770.	.736	2.376	126.89
767.28	184500.	33.93	4.72	7.99	6805.	.736	2.376	126.89
767.31	192100.	40.36	4.72	7.99	4000.	.870	2.466	113.11
767.32	192100.	40.36	4.72	7.99	4020.	.870	2.466	113.11
767.33	191700.	40.36	4.72	7.99	4230.	.855	2.466	115.08
767.34	191700.	40.36	4.72	7.99	4260.	.855	2.466	115.08
767.35	196900.	40.36	4.72	7.99	5740.	.775	2.466	126.89
767.36	196500.	40.36	4.72	7.99	5760.	.775	2.466	126.89
767.37	198200.	40.36	4.72	7.99	6060.	.800	2.487	131.15
777.1	298900.	48.97	5.87	8.68	3970.	.879	2.529	109.88
777.2	298900.	48.97	5.87	8.68	4240.	.864	2.529	111.83
777.3	299550.	48.97	5.87	8.68	4820.	.832	2.529	116.18
777.4	304500.	48.97	5.87	8.68	6030.	.793	2.540	125.95

```
777.5 304500. 48.97 5.87 8.68 6300. .780 2.540 128.12

# EW(lb) ILF(m) IWF(m) AR(-) RA(nm) MWLG(-) CLM(-) W/S(lb/ft2)
variables # 1 2 3 4 5 6 7
!!! TWIN engines ONLY

-* Solutions *-

C(0)=    6556.45
C(  1)=   0.753794
C(  2)=    1.07000
C(  3)=   0.297701
C(  4)=   0.244038
C(  5)=  -0.343887
C(  6)=   0.276215
C(  7)=  -0.847460

Aircraft type   |       % difference
    300.610     |         0.133909
    300.620     |         0.283507
    300.630     |          8.26092E-02
    300.640     |        -0.104994
    310.210     |         0.366016
    310.220     |         0.344582
    310.310     |        -0.949508
    310.320     |        -0.328149
    757.100     |          1.38342
    757.200     |        -0.200087
    757.300     |         0.426012
```

757.400	-0.947409
767.210	1.34884
767.220	0.392465
767.230	-0.615847
767.240	-1.48172
767.250	-0.312814
767.260	0.173701
767.270	1.01123
767.280	0.779141
767.310	-0.313029
767.320	-0.435198
767.330	-1.02537
767.340	-1.19975
767.350	-0.896331
767.360	-1.18759
767.370	2.07593
777.100	0.905445
777.200	0.202524
777.300	-0.775726
777.400	0.471234
777.500	0.282245

Mean error (%) = 0.669885

Appendix E

Conceptual Design of a Transport Aircraft

E.1 Computer Program : synthesis.f

```
PROGRAM SYNTHESIS

C written by FRUYTIER Pierre-Andre (1995)

C Synthesis of current twin turbofan transport aircraft :
C - single deck,
C - mid and long range,
C - mid and large capacity,
C - conventional tail and wing mounted engines
C at the conceptual design level

IMPLICIT NONE

REAL PI, ! pi constant (-)
```

```
& G, ! gravity acceleration (ft/s2)
& RHO, ! air specific mass @ sea level (sl/ft3)
& E, ! Oswald number (T/W) (-)
& EC, ! clean Oswald number (-)
& TV, ! V coefficient for thrust @ sea level (lb*s/ft)
& TV2, ! V2 coefficient for thrust @ sea level (lb*s2/ft2)
& TOGW, ! weight fraction after take-off (-)
& CDOC, ! CD0 clean (-)
& TSFCCR, ! TSFC @ cruise (1/hours)
& EN, ! endurance (hours)
& TSFCL, ! TSFC @ loiter (1/hours)
& D ! deceleration factor @ landing (-)

PARAMETER (PI=3.141592,
& G=32.16,
& RHO=.0023769,
& E=.7,
& EC=.8,
& TV=-.949E-3,
& TV2=.773E-6,
& TOGW=.97,
& CDOC=.016,
& TSFCCR=.5,
& EN=.75,
& TSFCL=.4,
& D=.34)

C Designer's seeds
```

```
INTEGER NSAE ! number of seat abreast - economy (-)

REAL AR, ! aspect ratio (-)
& CLM ! CL-maximum (-)

C Requirements

INTEGER NF, ! number of passengers - 1st class (-)
& NE ! number of passengers - economy (-)

REAL HCR, ! cruise altitude (ft)
& MCR, ! maximum Mach # @ cruise (-)
& RA, ! range (nm)
& MLGW, ! maximum landing over gross weight (-)
& LFL, ! landing field length (ft)
& VA ! approach velocity (knots)

C model variables

INTEGER NSAF ! number of seat abreast in first class (-)

REAL LC, ! cabin length (ft)
& WC, ! cabin width (ft)
& LF, ! fuselage length (ft)
& WF, ! fuselage width (ft)
& V, ! velocity (ft/s)
& VSL, ! stall speed at landing (ft/s)
& VSTO, ! stall speed at takeoff (ft/s)
& CDCL, ! CD/CL (-)
```

```
& VCR ! optimal cruise speed (knots)


C Weight variables

REAL WS, ! W/S @ take-off (lb/ft2)
& TWMA, ! T/W @ missed approach (-)
& TWSSC, ! T/W @ second segment of climb (-)
& TW, ! T/W - maximum (-)
& WCR, ! weight ratio @ cruise (-)
& OEW, ! empty weight (lb)
& PCW, ! weight of the passengers and crew (lb)
& WL, ! weight ratio @ loiter (-)
& WFG, ! WF/WG (-)
& W ! gross weight (lb)


CHARACTER*8 DATA

PRINT *, 'Data file name :'
READ *, DATA

OPEN(UNIT=1,FILE=DATA,FORM='FORMATTED')

READ (1,*) AR,CLM,NF,NE,NSAE,HCR,MCR,RA,MLGW,LFL

C Cabin width
C from NSAE
```

```
IF (NSAE.LT.7) THEN
  WC=(NSAE*20.25+19)/12.
ELSE
    WC=(NSAE*20.25+2*19)/12
END IF

C Number of seat abreast in first class
C from WC

IF (NSAE.LT.7) THEN
  NSAF=NINT((12*WC-24)/24)
ELSE
    NSAF=NINT((12*WC-2*24)/24)
END IF

C Fuselage width
C from WC

WF=WC+6.6/12

C Cabin length
C from NF, NSAF, NE and NSAE

LC=.87+5.49*(NF/NSAF)+2.42*(NE/NSAE)+.135*(NF+NE)

C Fuselage length
C from LC

LF=LC+42.42
```

```
C Stall speed @ landing
C from LFL and D

VSL=SQRT((.6*LFL-50/.0524)/(.0524*1.3**2/2/.2/G+1.15**2/2/D/G))

C Approach speed from stall speed @ landing
C from VSL

VA=1.3*VSL/1.689

C Wing loading required for landing
C from VSL, CLM and MLGW

WS=.5*RHO*VSL**2*CLM/MLGW

C Static thrust over gross weight required for missed approach
C from VSL, CLM and MLGW

V=1.5*1.1*VSL
CDCL=(CDOC+.045+(CLM/(1.5*1.1)**2)**2/PI/AR/E)/(CLM/(1.5*1.1)**2)
TWMA=2*MLGW/(1+TV*V+TV2*V**2)*(CDCL+.021)

C Stall speed @ take-off from stall speed @ landing
C from VSL and MLGW

VSTO=VSL*SQRT(TOGW/MLGW/.8)

C Static thrust over gross weight required for second segment of climb
```

```
C from VSTO and CLM

V=1.2*VSTO
CDCL=(CDOC+.018+(.8*CLM/1.2**2)**2/PI/AR/E)/(.8*CLM/1.2**2)
TWSSC=2*TOGW/(1+TV*V+TV2*V**2)*(CDCL+.024)

C Static thrust over gross weight
C from TWSSC and TWMA

TW=MAX(TWSSC,TWMA)

C Operating empty weight
C from LC, WC, RA, MLGW, AR, CLM and WS

OEW=6556.45*
&    ((LC*.305)**.7538)*
&    ((WC*.305)**1.07)*
&    (RA**.244)*
&    (MLGW**-.344)*
&    (AR**.2977)*
&    (CLM**.2762)*
&    (WS**-.8475)

C Payload and crew weight
C from NF and NE

PCW=2*205+(NF/18+NE/33)*150+NF*225+NE*205

C Cruise speed
```

```
C from MCR and HCR

IF (HCR.LE.36089) THEN
  VCR=MCR*1116.4*SQRT(1-6.875E-6*HCR)
ELSE
    VCR=MCR*968.1
END IF
VCR=VCR/1.689

C Cruise weight fraction
C from AR, RA, VCR

CDCL=SQRT(16*CD0C/(3*PI*AR*EC))
RA=RA+200
WCR=(EXP(-TSFCCR*RA*CDCL/VCR)+
&      (1-TSFCCR*RA*CDCL/(2*VCR))**2)/2

C Loiter weight fraction
C from AR

CDCL=SQRT(4*CD0C/(PI*AR*EC))
WL=EXP(-TSFCL*EN*CDCL)

C Fuel weight fraction
C from WCR and WL

WFG=1.06*(1-.99*.99*.995*.98*WCR*.99*WL*.992)

C Gross weight
```

```
C from OEW, PCW and WFG

W=OEW+PCW+WFG*(PCW+OEW)/(1-WFG)

PRINT*,'Number of seats abreast in first class (-) =',NSAF
PRINT*,'Fuselage length (ft) =',LF
PRINT*,'Fuselage width (ft) =',WF
PRINT*,'Cabin length (ft) =',LC
PRINT*,'Cabin width (ft) =',WC
PRINT*,'Approach velocity (knots) ='.VA
PRINT*,'Cruise velocity (knots) =',VCR
PRINT*,'W/S (lb/ft^2) =',WS
PRINT*,'T/W (-) =',TW
PRINT*,'S (ft^2) =',W/WS
PRINT*,'T (lb) =', TW*W/2
PRINT*,'W (lb) =',W
PRINT*,'Operating empty weight (lb) =',OEW
PRINT*,'Crew & payload weight (lb) =',PCW
PRINT*,'Fuel weight =',WFG*W
PRINT*,'Span (ft)=',SQRT(AR*W/WS)

END
```

E.2 File : TU204

```
9.67 3.366 12. 184. 6. 38050. .78 3415. .808 4411.
AR CLM NP1 NP2 NSA2 HCR MCR RA MWLG LFL

actual values :
```

```
Fuselage length (ft) = 150 ft 11 in
Fuselage diameter (ft) = 12 ft 5.5 in
Cabin length (ft) = 99 ft
Cabin width (ft) = 11 ft 8.5 in
Approach velocity (knots) = 122
Cruise velocity (knots) = 448
W/S (lb/ft^2) = 124.4
T/W (-) = .291 - 0.380
S (ft^2) = 1963.4
T (lb) = 35580 - 43100
max W (lb) = 244155
Empty weight (lb) = 130070
max Crew & payload weight (lb) = 43132
max Fuel weight = 72090
Span (ft)=    137 ft 9.5 in

approximated values (with SYNTHESIS.f) :

Number of seats abreast in first class (-) =  5
Fuselage length (ft) =    153.330
Fuselage width (ft) =    12.2583
Cabin length (ft) =    110.910
Cabin width (ft) =    11.7083
Approach velocity (knots) =    122.003
Cruise velocity (knots) =    447.080
W/S (lb/ft^2) =    124.393
T/W (-) =   0.307524
S (ft^2) =    1972.21
```

```
T (lb) =    37722.3
W (lb) =    245329.
Operating empty weight (lb) =    131381.
Crew & payload weight (lb) =    41580.0
Fuel weight =    72368.6
Span (ft)=    138.099
```

Appendix F

Conceptual Design Optimization

F.1 Computer Program : thesis.f

```
PROGRAM THESIS

C written by FRUYTIER Pierre-Andre (1994)

C Conceptual Design Optimization of Twin Turbofan Transport Aircraft

IMPLICIT NONE

INTEGER N

REAL PRECISION,CROSSOVERORMUTATION

PARAMETER (N=30) ! even integer only

PARAMETER (PRECISION=.01,CROSSOVERORMUTATION=.5)
```

```
INTEGER SEEDS,START,P,I
INTEGER CODELENGTH_NSA2,CODELENGTH_AR,CODELENGTH_CLM
INTEGER POSITION,CO,M

REAL NEWFITNESS

INTEGER*4 CODELENGTHFORINTEGER,CODELENGTHFORREAL
REAL SECNDS
REAL RAN
REAL FITNESSEVALUATION

REAL FITNESS(30)

INTEGER ADDRESS(30)

INTEGER NSA2I,NSA2S
REAL ARI,ARS
REAL CLMI,CLMS

CHARACTER GO

STRUCTURE /REQUIREMENT/
INTEGER NP1
INTEGER NP2
REAL HCR
REAL MCR
REAL VCR
REAL RA
REAL MWLG
```

```
REAL VA
REAL LFL
REAL MCF
REAL CLAR
END STRUCTURE

STRUCTURE /AIRCRAFT/
REAL W
REAL EW
REAL FW
REAL TW
REAL WS
CHARACTER*10 AR_CODE
REAL AR
CHARACTER*10 CLM_CODE
REAL CLM
CHARACTER*10 NSA2_CODE
INTEGER NSA2
REAL IWF
REAL ILF
REAL LDCL
REAL LDCR
REAL LDLO
END STRUCTURE

RECORD /REQUIREMENT/ R

RECORD /AIRCRAFT/ AC(30),NEW,MUTANT,GUY1,GUY2
```

```
OPEN(UNIT=1,FILE='REQUIREMENT.DAT',FORM='FORMATTED')
READ (1,*) NSA2I,NSA2S
READ (1,*) ARI,ARS
READ (1,*) CLMI,CLMS
READ (1,*) R.NP1,R.NP2
READ (1,*) R.HCR,R.MCR,R.RA
READ (1,*) R.MWLG,R.LFL
READ (1,*) R.MCF
READ (1,*) R.CLAR
CLOSE(1)

PRINT*,'*** Requirements ***'
PRINT*,''
PRINT*,' Range for the genes :'
PRINT 1, NSA2I,NSA2S
PRINT 2, ARI,ARS
PRINT 3, CLMI,CLMS
PRINT*,''
PRINT*,' Parameters :'
PRINT 4, R.NP1
PRINT 5, R.NP2
PRINT 6, R.HCR
PRINT 7, R.MCR
PRINT 8, R.RA
PRINT 9, R.MWLG
PRINT 10, R.LFL
PRINT*,''

SEEDS=NINT(SECNDS(0.)*10+1)
```

```
CODELENGTH_NSA2=CODELENGTHFORINTEGER(NSA2I,NSA2S)
CODELENGTH_AR=CODELENGTHFORREAL(ARI,ARS,PRECISION)
CODELENGTH_CLM=CODELENGTHFORREAL(CLMI,CLMS,PRECISION)
CALL PRESELECTION(N,FITNESS,ADDRESS,START)

DO P=1,N
  NEW.NSA2=NSA2S+1
  DO WHILE (NEW.NSA2 .GT. NSA2S)
    CALL RANDOMCODE(CODELENGTH_NSA2,NEW.NSA2_CODE,SEEDS)
    CALL GRAY2INTEGER
& (CODELENGTH_NSA2,NEW.NSA2_CODE,NSA2I,NEW.NSA2)
  END DO
  NEW.AR=ARS+1
  DO WHILE (NEW.AR .GT. ARS)
    CALL RANDOMCODE(CODELENGTH_AR,NEW.AR_CODE,SEEDS)
    CALL GRAY2REAL
& (CODELENGTH_AR,NEW.AR_CODE,ARI,PRECISION,NEW.AR)
  END DO
  NEW.CLM=CLMS+1
  DO WHILE (NEW.CLM .GT. CLMS)
    CALL RANDOMCODE(CODELENGTH_CLM,NEW.CLM_CODE,SEEDS)
    CALL GRAY2REAL
& (CODELENGTH_CLM,NEW.CLM_CODE,CLMI,PRECISION,NEW.CLM)
  END DO
  NEWFITNESS=FITNESSEVALUATION(R,NEW)
  CALL SELECTION(NEWFITNESS,FITNESS,START,ADDRESS,POSITION)
  CALL COPYRECORD(AC(POSITION),NEW)
END DO
```

```
PRINT 11, R.VCR
PRINT 12, R.VA
PRINT*,''
PRINT*,'Manufacturing :'
PRINT 13, R.MCF
PRINT 14, R.CLAR
PRINT*,''

CALL PRINTGENERATION(0,START,ADDRESS,FITNESS,AC)

PRINT *, 'Do you want to initiate the Genetic Algorithm [y/n] ?'
READ*,GO
PRINT *,''
I=1
DO WHILE (GO.NE.'n')
  DO CO=1,N/2
    IF (RAN(SEEDS) .LE. CROSSOVERORMUTATION) THEN
      DO M=2*CO-1,2*CO
CALL COPYRECORD(MUTANT,AC(M))

MUTANT.NSA2=NSA2S+1
DO WHILE (MUTANT.NSA2 .GT. NSA2S)
  CALL MUTATION1BIT(CODELENGTH_NSA2,MUTANT.NSA2_CODE,SEEDS)
  CALL GRAY2INTEGER
&    (CODELENGTH_NSA2,MUTANT.NSA2_CODE,NSA2I,MUTANT.NSA2)
END DO
MUTANT.AR=ARS+1
DO WHILE (MUTANT.AR .GT. ARS)
  CALL MUTATION1BIT(CODELENGTH_AR,MUTANT.AR_CODE,SEEDS)
```

```
  CALL GRAY2REAL
&    (CODELENGTH_AR,MUTANT.AR_CODE,ARI,PRECISION,MUTANT.AR)
END DO
MUTANT.CLM=CLMS+1
DO WHILE (MUTANT.CLM .GT. CLMS)
  CALL RANDOMCODE(CODELENGTH_CLM,MUTANT.CLM_CODE,SEEDS)
  CALL GRAY2REAL
&           (CODELENGTH_CLM,MUTANT.CLM_CODE,CLMI,PRECISION,MUTANT.CLM)
END DO

NEWFITNESS=FITNESSEVALUATION(R,MUTANT)
CALL SELECTION(NEWFITNESS,FITNESS,START,ADDRESS,POSITION)
IF (POSITION .NE. 0) THEN
  CALL COPYRECORD(AC(POSITION),MUTANT)
END IF
      END DO
    ELSE
CALL COPYRECORD(GUY1,AC(2*CO-1))
CALL COPYRECORD(GUY2,AC(2*CO))

GUY1.NSA2=NSA2S+1
GUY2.NSA2=NSA2S+1
DO WHILE
& ((GUY1.NSA2 .GT. NSA2S).OR.(GUY2.NSA2 .GT. NSA2S))
  CALL CROSSOVERMASK
&         (CODELENGTH_NSA2,GUY1.NSA2_CODE,GUY2.NSA2_CODE,SEEDS)
  CALL GRAY2INTEGER
&         (CODELENGTH_NSA2,GUY1.NSA2_CODE,NSA2I,GUY1.NSA2)
  CALL GRAY2INTEGER
```

```
&          (CODELENGTH_NSA2,GUY2.NSA2_CODE,NSA2I,GUY2.NSA2)
END DO
GUY1.AR=ARS+1
GUY2.AR=ARS+1
DO WHILE
& ((GUY1.AR .GT. ARS).OR.(GUY2.AR .GT. ARS))
  CALL CROSSOVERMASK
&          (CODELENGTH_AR,GUY1.AR_CODE,GUY2.AR_CODE,SEEDS)
  CALL GRAY2REAL
&          (CODELENGTH_AR,GUY1.AR_CODE,ARI,PRECISION,GUY1.AR)
  CALL GRAY2REAL
&          (CODELENGTH_AR,GUY2.AR_CODE,ARI,PRECISION,GUY2.AR)
END DO
GUY1.CLM=CLMS+1
GUY2.CLM=CLMS+1
DO WHILE
& ((GUY1.CLM .GT. CLMS).OR.(GUY2.CLM .GT. CLMS))
  CALL CROSSOVERMASK
&          (CODELENGTH_CLM,GUY1.CLM_CODE,GUY2.CLM_CODE,SEEDS)
  CALL GRAY2REAL
&          (CODELENGTH_CLM,GUY1.CLM_CODE,CLMI,PRECISION,GUY1.CLM)
  CALL GRAY2REAL
&          (CODELENGTH_CLM,GUY2.CLM_CODE,CLMI,PRECISION,GUY2.CLM)
END DO

NEWFITNESS=FITNESSEVALUATION(R,GUY1)
CALL SELECTION(NEWFITNESS,FITNESS,START,ADDRESS,POSITION)
IF (POSITION .NE. 0) THEN
  CALL COPYRECORD(AC(POSITION),GUY1)
```

```
END IF
NEWFITNESS=FITNESSEVALUATION(R,GUY2)
CALL SELECTION(NEWFITNESS,FITNESS,START,ADDRESS,POSITION)
IF (POSITION .NE. 0) THEN
  CALL COPYRECORD(AC(POSITION),GUY2)
END IF
    END IF
  END DO
  CALL PRINTGENERATION(I,START,ADDRESS,FITNESS,AC)
  PRINT*,'Do you want another generation [y/n] ?'
  READ*,GO
  PRINT*,''
  I=I+1
END DO

1 FORMAT('Seats Abreast # for 2nd class [',I1,',',I2,']')
2 FORMAT('Aspect Ratio [',F4.2,',',F4.2,']')
3 FORMAT('Maximum Lift Coefficient [',F5.3,',',F5.3,']')
4 FORMAT('Passengers # : 1st class : ',I2)
5 FORMAT('              2nd class : ',I3)
6 FORMAT('Cruise altitude : ',F6.0,' ft')
7 FORMAT('Cruise Mach # : ',F4.2)
8 FORMAT('Range : ',F5.0,' nm')
9 FORMAT('Landing weight/gross weight : ',F4.2)
10 FORMAT('Landing Field Length : ',F5.0,' ft')
11 FORMAT('Cruise velocity : ',F4.0,' knots')
12 FORMAT('Approach velocity : ',F4.0,' knots')
13 FORMAT('Maximum cabin finess : ',F5.2)
14 FORMAT('Maximum CLmax*AR : ',F5.2)
```

```
END

INCLUDE 'codes.f'
INCLUDE 'GA.f'
INCLUDE 'aircraft.f'
```

F.2 Library of Subroutines : GA.f

```
C written by FRUYTIER Pierre-Andre (1994)

SUBROUTINE CROSSOVER1POINT
& (CODELENGTH,PARENT1CHILD,PARENT2CHILD,SEEDS)

IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) PARENT1CHILD
CHARACTER*(*) PARENT2CHILD
INTEGER SEEDS

INTEGER POINT,BIT
REAL RAN

POINT=NINT(AINT((CODELENGTH*RAN(SEEDS))))+1
DO BIT=POINT,CODELENGTH
  PARENT1CHILD(BIT:BIT)=PARENT2CHILD(BIT:BIT)
  PARENT2CHILD(BIT:BIT)=PARENT1CHILD(BIT:BIT)
```

```
END DO

RETURN

END

SUBROUTINE CROSSOVERMASK
& (CODELENGTH,PARENT1CHILD,PARENT2CHILD,SEEDS)

IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) PARENT1CHILD
CHARACTER*(*) PARENT2CHILD
INTEGER SEEDS

INTEGER BIT
REAL RAN

DO BIT=1,CODELENGTH
  IF (RAN(SEEDS) .GE. 0.5) THEN
    PARENT1CHILD(BIT:BIT)=PARENT2CHILD(BIT:BIT)
    PARENT2CHILD(BIT:BIT)=PARENT1CHILD(BIT:BIT)
  END IF
END DO

RETURN

END
```

```
INTEGER FUNCTION CODELENGTHFORINTEGER
& (LOWERBOUND,UPPERBOUND)

IMPLICIT NONE

INTEGER LOWERBOUND
INTEGER UPPERBOUND

INTEGER CODELENGTH

CODELENGTH=NINT(AINT(LOG(UPPERBOUND-LOWERBOUND+1.)/LOG(2.)))
CODELENGTHFORINTEGER=CODELENGTH+1

RETURN

END

INTEGER FUNCTION CODELENGTHFORREAL
& (LOWERBOUND,UPPERBOUND,PRECISION)

IMPLICIT NONE

REAL LOWERBOUND
REAL UPPERBOUND
REAL PRECISION

REAL STEPNUMBER
INTEGER CODELENGTH
```

```
STEPNUMBER=LOG(UPPERBOUND/LOWERBOUND)/LOG(1+PRECISION)
CODELENGTH=NINT(AINT(LOG(STEPNUMBER)/LOG(2.)))
CODELENGTHFORREAL=CODELENGTH+1

RETURN

END


SUBROUTINE MUTATION1BIT
& (CODELENGTH,CODE,SEEDS)

IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) CODE
INTEGER SEEDS

INTEGER BITTOCHANGE
REAL RAN

BITTOCHANGE=NINT(AINT((CODELENGTH*RAN(SEEDS))))+1
IF (CODE(BITTOCHANGE:BITTOCHANGE) .EQ. '1') THEN
  CODE(BITTOCHANGE:BITTOCHANGE)='0'
ELSE
    CODE(BITTOCHANGE:BITTOCHANGE)='1'
END IF

RETURN
```

```
END

SUBROUTINE PRESELECTION
& (MEMBERSNUMBER,FITNESS,ADDRESS,START)

IMPLICIT NONE

INTEGER MEMBERSNUMBER
REAL FITNESS(*)
INTEGER ADDRESS(*)
INTEGER START

INTEGER NUMBER

DO NUMBER=1,MEMBERSNUMBER
  FITNESS(NUMBER)=0.
  ADDRESS(NUMBER)=NUMBER+1
END DO
START=1
ADDRESS(MEMBERSNUMBER)=0

RETURN

END

SUBROUTINE RANDOMCODE
& (CODELENGTH,CODE,SEEDS)
```

```
IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) CODE
INTEGER*4 SEEDS

INTEGER BIT
REAL RAN

DO BIT=1,CODELENGTH
  IF (RAN(SEEDS) .LE. 0.5) THEN
    CODE(BIT:BIT)='0'
  ELSE
      CODE(BIT:BIT)='1'
  END IF
END DO

RETURN

END

SUBROUTINE SELECTION
& (NEWFITNESS,FITNESS,START,ADDRESS,POSITION)

IMPLICIT NONE

REAL NEWFITNESS
REAL FITNESS(*)
INTEGER START
```

```
INTEGER ADDRESS(*)
INTEGER POSITION

INTEGER NEXTONE,LASTONE,NEWSTART
LOGICAL SEARCHING

NEXTONE=START
SEARCHING=.TRUE.
DO WHILE ((NEXTONE .NE. 0) .AND. (SEARCHING))
  IF ((FITNESS(NEXTONE) .EQ. 0) .OR.
&      (FITNESS(NEXTONE) .GE. NEWFITNESS)) THEN
    LASTONE=NEXTONE
    NEXTONE=ADDRESS(LASTONE)
  ELSE
      SEARCHING=.FALSE.
  END IF
END DO
IF (NEXTONE .NE. START) THEN
  FITNESS(START)=NEWFITNESS
  POSITION=START
  IF (LASTONE .NE. START) THEN
    NEWSTART=ADDRESS(START)
    ADDRESS(LASTONE)=START
    ADDRESS(START)=NEXTONE
    START=NEWSTART
  END IF
ELSE
    POSITION=0
END IF
```

```
RETURN

END
```

F.3 Library of Subroutines : codes.f

```
C written by FRUYTIER Pierre-Andre (1994)

SUBROUTINE BINARY2INTEGER
& (CODELENGTH,BINARYCODE,LOWERBOUND,INTEGERVALUE)

IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) BINARYCODE
INTEGER LOWERBOUND
INTEGER INTEGERVALUE

INTEGER NATURALVALUE,BIT

NATURALVALUE=0
DO BIT=1,CODELENGTH
  IF (BINARYCODE(BIT:BIT) .EQ. '1') THEN
  NATURALVALUE=NATURALVALUE+2**(BIT-1)
  END IF
END DO
```

```
INTEGERVALUE=LOWERBOUND+NATURALVALUE

RETURN

END

SUBROUTINE BINARY2REAL
& (CODELENGTH,BINARYCODE,LOWERBOUND,PRECISION,REALVALUE)

IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) BINARYCODE
REAL LOWERBOUND
REAL PRECISION
REAL REALVALUE

INTEGER NATURALVALUE,BIT

NATURALVALUE=0
DO BIT=1,CODELENGTH
  IF (BINARYCODE(BIT:BIT) .EQ. '1') THEN
  NATURALVALUE=NATURALVALUE+2**(BIT-1)
  END IF
END DO

REALVALUE=LOWERBOUND*(1+PRECISION)**NATURALVALUE

RETURN
```

```
& (CODELENGTH,GRAYCODE,LOWERBOUND,PRECISION,REALVALUE)

IMPLICIT NONE

INTEGER CODELENGTH
CHARACTER*(*) GRAYCODE
REAL LOWERBOUND
REAL PRECISION
REAL REALVALUE

INTEGER NATURALVALUE,BIT

NATURALVALUE=0
DO BIT=1,CODELENGTH
  IF (GRAYCODE(BIT:BIT).EQ.'1') THEN
  NATURALVALUE=2**BIT-1-NATURALVALUE
  END IF
END DO

REALVALUE=LOWERBOUND*(1+PRECISION)**NATURALVALUE

RETURN

END

SUBROUTINE INTEGER2BINARY
& (CODELENGTH,INTEGERVALUE,LOWERBOUND,BINARYCODE)

IMPLICIT NONE
```

```
INTEGER CODELENGTH
INTEGER INTEGERVALUE
INTEGER LOWERBOUND
CHARACTER*(*) BINARYCODE

INTEGER NATURALVALUE,BIT,ODDOREVEN

NATURALVALUE=INTEGERVALUE-LOWERBOUND

DO BIT=1,CODELENGTH
  ODDOREVEN=NATURALVALUE/(2**(BIT-1))
  IF (NATURALVALUE .LT. 2**(BIT-1)) THEN
    BINARYCODE(BIT:BIT)='0'
  ELSE IF (MOD(ODDOREVEN,2) .EQ. 0) THEN
 BINARYCODE(BIT:BIT)='0'
       ELSE
   BINARYCODE(BIT:BIT)='1'
  END IF
END DO

RETURN

END

SUBROUTINE INTEGER2GRAY
& (CODELENGTH,INTEGERVALUE,LOWERBOUND,GRAYCODE)

IMPLICIT NONE
```

```
INTEGER CODELENGTH
INTEGER INTEGERVALUE
INTEGER LOWERBOUND
CHARACTER*(*) GRAYCODE

INTEGER NATURALVALUE,BIT,ODDOREVEN

NATURALVALUE=INTEGERVALUE-LOWERBOUND

DO BIT=1,CODELENGTH
  ODDOREVEN=(NATURALVALUE-2**(BIT-1))/(2**BIT)
  IF (NATURALVALUE .LT. 2**(BIT-1)) THEN
    GRAYCODE(BIT:BIT)='0'
  ELSE IF (MOD(ODDOREVEN,2) .EQ. 0) THEN
 GRAYCODE(BIT:BIT)='1'
       ELSE
   GRAYCODE(BIT:BIT)='0'
  END IF
END DO

RETURN

END

SUBROUTINE REAL2BINARY
& (CODELENGTH,REALVALUE,LOWERBOUND,PRECISION,BINARYCODE)

IMPLICIT NONE
```

```
INTEGER CODELENGTH
REAL REALVALUE
REAL LOWERBOUND
REAL PRECISION
CHARACTER*(*) BINARYCODE

INTEGER NATURALVALUE,BIT,ODDOREVEN

NATURALVALUE=NINT(LOG(REALVALUE/LOWERBOUND)/LOG(1+PRECISION))

DO BIT=1,CODELENGTH
  ODDOREVEN=NATURALVALUE/(2**(BIT-1))
  IF (NATURALVALUE .LT. 2**(BIT-1)) THEN
    BINARYCODE(BIT:BIT)='0'
  ELSE IF (MOD(ODDOREVEN,2) .EQ. 0) THEN
 BINARYCODE(BIT:BIT)='0'
       ELSE
   BINARYCODE(BIT:BIT)='1'
  END IF
END DO

RETURN

END

SUBROUTINE REAL2GRAY
& (CODELENGTH,REALVALUE,LOWERBOUND,PRECISION,GRAYCODE)
```

```
IMPLICIT NONE

INTEGER CODELENGTH
REAL REALVALUE
REAL LOWERBOUND
REAL PRECISION
CHARACTER*(*) GRAYCODE

INTEGER NATURALVALUE,BIT,ODDOREVEN

NATURALVALUE=NINT(LOG(REALVALUE/LOWERBOUND)/LOG(1+PRECISION))

DO BIT=1,CODELENGTH
  ODDOREVEN=(NATURALVALUE-2**(BIT-1))/(2**BIT)
  IF (NATURALVALUE .LT. 2**(BIT-1)) THEN
    GRAYCODE(BIT:BIT)='0'
  ELSE IF (MOD(ODDOREVEN,2) .EQ. 0) THEN
 GRAYCODE(BIT:BIT)='1'
       ELSE
   GRAYCODE(BIT:BIT)='0'
  END IF
END DO

RETURN

END
```

F.4 Library of Subroutines : aircraft.f

```
C written by FRUYTIER Pierre-Andre (1994)

REAL FUNCTION FITNESSEVALUATION (R,AC)

REAL PI, ! pi constant (-)
& G, ! gravity acceleration (ft/s2)
& RHO, ! air specific mass @ sea level (sl/ft3)
& E, ! Oswald number (-)
& TV, ! V coefficient for thrust @ sea level (lb*s/ft)
& TV2, ! V2 coefficient for thrust @ sea level (lb*s2/ft2)
& C, ! acceleration @ climb (s2/ft2)
& CDOC, ! CDO clean (-)
& TSFCCR, ! TSFC @ cruise (1/hours)
& EN, ! endurance (hours)
& TSFCL, ! TSFC @ loiter (1/hours)
& D ! deceleration @ landing (ft/sec2)

PARAMETER (PI=3.141592,
& G=32.16,
& RHO=.0023769,
& E=.7,
& TV=-1.00769E-3,
& TV2=6.6656E-7,
& C=4.549236241E-7,
& CDOC=.016,
& TSFCCR=.5,
```

```
& EN=.75,
& TSFCL=.4,
& D=12.0178)

INTEGER NSA1 ! number of seat abreast - 1st class (-)

REAL V, ! velocity (ft/s)
& WCR, ! weight ratio @ cruise (-)
& WL, ! weight ratio @ loiter (-)
& WFG, ! WF/WG (-)
& PCW ! passengers & crew weight (lb)

REAL FITNESSEVALUATION

STRUCTURE /REQUIREMENT/
INTEGER NP1
INTEGER NP2
REAL HCR
REAL MCR
REAL VCR
REAL RA
REAL MWLG
REAL VA
REAL LFL
REAL MCF
REAL CLAR
END STRUCTURE

STRUCTURE /AIRCRAFT/
```

```
REAL W
REAL EW
REAL FW
REAL TW
REAL WS
CHARACTER*10 AR_CODE
REAL AR
CHARACTER*10 CLM_CODE
REAL CLM
CHARACTER*10 NSA2_CODE
INTEGER NSA2
REAL IWF
REAL ILF
REAL LDCL
REAL LDCR
REAL LDLO
END STRUCTURE

RECORD /REQUIREMENT/ R

RECORD /AIRCRAFT/ AC

C Seating arrangement -> IWF & ILF

IF (AC.NSA2.LT.7) THEN
  AC.IWF=(AC.NSA2*20+19)/12.
  NSA1=NINT((AC.IWF*12-24)/24)
ELSE
    AC.IWF=(AC.NSA2*20+2*19)/12.
```

```
    NSA1=NINT((AC.IWF*12-2*19)/24)
END IF
AC.ILF=((R.NP1/NSA1)*39+40+(R.NP2/AC.NSA2)*35
& +(R.NP1/15+R.NP2/50)*40)/12.

C Landing -> Vstall @ landing & Vapproach (knots)

V=SQRT(2*D*(R.LFL*.6-50/.0524)/(1.15**2+1.3**2*D*.0524/.2/G))
R.VA=1.3*V/1.689

C Landing vs Take-off -> Vstall @ take-off

V=V*SQRT(.99*.99*.995/R.MWLG/.8)

C Take-off -> WS

AC.WS=.5*RHO*V**2*AC.CLM*.8/(.99*.99*.995)

C Second segment of climb -> TW

V=1.2*V
AC.LDCL=(.8*AC.CLM/1.2**2)/
& (CDOC+.02+(.8*AC.CLM/1.2**2)**2/PI/AC.AR/.95/E)
AC.TW=2*(.99*.99*.995)/(1+TV*V+TV2*V**2)
&        *(1/AC.LDCL+.024*(1+C*V**2))

C Cruise -> WCR

IF (R.HCR.LE.36089) THEN
```

```
  R.VCR=R.MCR*1115.6*SQRT(1-6.875E-6*R.HCR)
ELSE
    R.VCR=R.MCR*967.95
END IF
AC.LDCR=1/((SQRT(3.)+1/SQRT(3.))*SQRT(CD0C/(PI*AC.AR*E)))
WCR=(EXP(-(R.RA+200)*6076.*TSFCCR/3600./AC.LDCR/R.VCR)+
&      ((R.RA+200)*6076*TSFCCR/3600./AC.LDCR/R.VCR/2.-1)**2)/2
R.VCR=R.VCR/1.689

C Loiter -> WL

AC.LDLO=1/(2*SQRT(CD0C/(PI*AC.AR*E)))
WL=EXP(-EN*TSFCL/AC.LDLO)

C Weight analysis

WFG=1.06*(1-.99*.99*.995*.98*WCR*.99*WL*.992)
PCW=(R.NP1+R.NP2+R.NP1/18+R.NP2/34+2)*180.
& +(R.NP1+R.NP2)*50.

C model #1
C
C AC.EW=16073.5*
C&         ((AC.ILF*.305)**.992389)*
C&         ((AC.IWF*.305)**1.21866)*
C&         (AC.AR**.219539)*
C&         (R.RA**1.04075E-1)*
C&         (R.MWLG**-1.12187)*
C&         (AC.CLM**.447405)*
```

```
C&        (AC.WS**-1.04669)

C model #2
C
AC.EW=5990.73*
&        ((AC.ILF*.305)**.974528)*
&        ((AC.IWF*.305)**1.17431)*
&        (AC.AR**.225371)*
&        (R.RA**.201856)*
&        (R.MWLG**-.644954)*
&        (AC.CLM**.454614)*
&        (AC.WS**-.969587)

AC.FW=WFG*(PCW+AC.EW)/(1-WFG)
AC.W=PCW+AC.EW+AC.FW

FITNESSEVALUATION=AC.W
IF ((AC.ILF/AC.IWF).GT.(R.MCF)) THEN
  FITNESSEVALUATION=100000000.
END IF
IF ((AC.CLM*AC.AR).GT.(R.CLAR)) THEN
  FITNESSEVALUATION=100000000.
END IF

RETURN

END

SUBROUTINE COPYRECORD(OLDAC,NEWAC)
```

```
STRUCTURE /AIRCRAFT/
REAL W
REAL EW
REAL FW
REAL TW
REAL WS
CHARACTER*10 AR_CODE
REAL AR
CHARACTER*10 CLM_CODE
REAL CLM
CHARACTER*10 NSA2_CODE
INTEGER NSA2
REAL IWF
REAL ILF
REAL LDCL
REAL LDCR
REAL LDLO
END STRUCTURE

RECORD /AIRCRAFT/ OLDAC,NEWAC

OLDAC.W=NEWAC.W
OLDAC.EW=NEWAC.EW
OLDAC.FW=NEWAC.FW
OLDAC.TW=NEWAC.TW
OLDAC.WS=NEWAC.WS
OLDAC.AR_CODE=NEWAC.AR_CODE
OLDAC.AR=NEWAC.AR
```

```
OLDAC.CLM_CODE=NEWAC.CLM_CODE
OLDAC.CLM=NEWAC.CLM
OLDAC.NSA2_CODE=NEWAC.NSA2_CODE
OLDAC.NSA2=NEWAC.NSA2
OLDAC.IWF=NEWAC.IWF
OLDAC.ILF=NEWAC.ILF
OLDAC.LDCL=NEWAC.LDCL
OLDAC.LDCR=NEWAC.LDCR
OLDAC.LDLO=NEWAC.LDLO

RETURN

END

SUBROUTINE PRINTGENERATION (I,START,ADDRESS,FITNESS,AC)

INTEGER I,START,ADDRESS(*)

REAL FITNESS(*)

STRUCTURE /AIRCRAFT/
REAL W
REAL EW
REAL FW
REAL TW
REAL WS
CHARACTER*10 AR_CODE
REAL AR
CHARACTER*10 CLM_CODE
```

```
REAL CLM
CHARACTER*10 NSA2_CODE
INTEGER NSA2
REAL IWF
REAL ILF
REAL LDCL
REAL LDCR
REAL LDLO
END STRUCTURE

RECORD /AIRCRAFT/ AC(*)

PRINT*,'Generation #',I
PRINT*,'Fitness    W       EW      FW     T/W     W/S      T       S
       B        AR      CLM     NSA2    IWF     ILF
 L/D@cl  L/D@cr  L/D@lo'
PRINT*,'           (lb)    (lb)    (lb)    (-)    (lb/ft2)  (lb)    (ft2)
  (ft)     (-)     (-)     (-)     (ft)    (ft)    (-)     (-)     (-)'
CURSOR=START
DO WHILE (CURSOR .NE. 0)
  PRINT 1, FITNESS(CURSOR),AC(CURSOR).W,AC(CURSOR).EW,AC(CURSOR).FW,
&      AC(CURSOR).TW,AC(CURSOR).WS,
&      AC(CURSOR).TW*AC(CURSOR).W/2,AC(CURSOR).W/AC(CURSOR).WS,
&      SQRT(AC(CURSOR).AR*AC(CURSOR).W/AC(CURSOR).WS),
&      AC(CURSOR).AR,AC(CURSOR).CLM,AC(CURSOR).NSA2,
&      AC(CURSOR).IWF,AC(CURSOR).ILF,
&      AC(CURSOR).LDCL,AC(CURSOR).LDCR,AC(CURSOR).LDLO
  CURSOR=ADDRESS(CURSOR)
END DO
```

```
PRINT*,''

1 FORMAT(4F8.0,F6.3,F8.1,F9.0,F7.0,F9.2,F7.2,F8.3,4X,I2,2X,
&        F8.2,F9.2,F7.2,2F8.2)

RETURN

END
```

F.5 File : REQUIREMENT.DAT

```
6. 10.
7.7 9.4
2.1 3.1
18. 198.
39200. .8 3160.
.9 4806.
7.5
19.2.

requirement definition
NSAI NSAS
ARI ARS
CLMI CLMS
NP1 NP2
HCR MCR RA
MWLG LFL
MCF
MCLAR
```